Nose Tip Recession Measuring System for Hypersonic Test Vehicles

1977

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NOSE TIP RECESSION MEASURING SYSTEM
FOR HYPERSONIC TEST VEHICLES

BY

JAMES ANTHONY BROWN
B.S., University of Tennessee, 1968

RESEARCH REPORT

Submitted in partial fulfillment of the requirements for the degree of Master of Science in Engineering in the Graduate Studies Program of Florida Technological University

Orlando, Florida
1977
ABSTRACT

A method is presented which permits the measure of nose tip recession of re-entry vehicles and advanced terminal interceptors by employing a double choked flow coolant gas system. Recession of the tip results in an increased exit flow area which reduces the total pressure of the gas in the blast tube. Measurement of the blast tube pressure and gas generator (chamber) pressure will produce an effective measurement of the nose tip recession as long as choked flow (i.e., sonic velocity) is maintained in both the tip exit area and the gas generator throat area.

Governing flow equations documented in the literature are developed for double choked flow. Hypersonic wind tunnel test data are presented to verify the developed flow equations and to identify the mass flow ratios necessary to sustain double choked flow.
ACKNOWLEDGEMENTS

Sincere thanks are extended to my research paper director, Dr. Ronald D. Evans. His encouragement and guidance have been invaluable in completing this project. The time donated by the other members of my committee, Dr. Michael Varney and Mr. James Beck, is also greatly appreciated.

Special thanks are given to my wife, Cindy, who shared in this effort by typing all the drafts and the final copy of this research report.
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LIST OF SYMBOLS

\( A \) = area, \( \text{ft}^2 \)

\( C_D' \) = discharge coefficient

\( C_D \) = non-dimensional drag coefficient

\( C_P \) = specific heat at constant pressure, \( \text{BTU/lb} \cdot ^\circ\text{R} \)

\( C_V \) = specific heat at constant volume, \( \text{BTU/lb} \cdot ^\circ\text{R} \)

\( D_{\text{TIP}} \) = GASJET tip exit diameter, inches

\( g \) = acceleration due to gravity, \( 32.2 \text{ ft/sec}^2 \)

\( g_c \) = constant, \( 32.2 \text{ lbm ft/lb sec}^2 \)

\( K = \frac{C_D' \left( \frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{\gamma-1}}}{\gamma g_c \left( \frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{\gamma-1}} \cdot \text{lbm(}^\circ\text{R})^{1/2} / \text{lbf sec} \)

\( M \) = mach number

\( \dot{M} \) = mass flow rate, \( \text{lb/sec} \)

\( \dot{m} \) = mass flow ratio

\( P \) = wind tunnel ambient pressure, \( \text{lb/in}^2 \)

\( P_{\text{BT}} \) = pressure measured in GASJET blast tube, \( \text{lb/in}^2 \)

\( P_{\text{T1}} \) = pressure measured at GASJET tip exit, \( \text{lb/in}^2 \)

\( P_{\text{sup}} \) = supply pressure = \( P_c \), \( \text{lb/in}^2 \)

\( P_{T2} \) = total pressure behind shock, \( \text{lb/in}^2 \)

\( P_T \) = total pressure, \( \text{lb/in}^2 \)
R = gas constant, ft-lbf/lbm
r = recession, inches
\( T_T \) = total temperature, \(^\circ\)R
W = weight, lbf
\( W_a \) = weight flow rate, lbf/sec
\( \gamma \) = ratio of specific heats \((C_p/C_v)\)
\( \Pi \) = numerical constant
\( \Theta \) = convergence half-angle of tip flow passage, degrees
I. INTRODUCTION

Re-entry can be defined as the return of a spacecraft or ballistic missile into the earth's atmosphere. During re-entry, the relatively dense gas surrounding the earth provides a braking force resulting from aerodynamic drag. At the same time, it dissipates the high kinetic energy of returning vehicles in the form of heat.

The parameter, \( \frac{W}{C_D A} \), called the ballistic coefficient, is used as a basis for comparison of re-entry performance. In this parameter, \( W \) is the weight of the vehicle, \( C_D \) is a non dimensional drag coefficient for the shape used and \( A \) is a representative cross sectional area (typically the projected area of the vehicle). Blunt shapes such as the Apollo spacecraft have a characteristically low value of the ballistic coefficient because of relatively high values of \( C_D \) and \( A \). Typical low ballistic coefficient values are 100 to 150 lb/ft\(^2\). Typical high values are 1000 lb/ft\(^2\) or above. Figure 1 illustrates how a re-entry vehicle with a high ballistic coefficient...
plunges farther into the atmosphere than one with a low ballistic coefficient before the velocity is decreased. This in turn means that the maximum deceleration is reached at a lower altitude where the atmosphere is denser. For the returning spacecraft, one would want a low ballistic coefficient which would dissipate the high kinetic energy at a higher, less dense portion of the atmosphere. For the ballistic missile, one would want a high ballistic coefficient which corresponds to relatively low $C_D$ values. Keeping the aerodynamic drag low for such a vehicle allows for quicker penetration of the atmosphere which reduces the delivery time to a potential target, reduces the "loiter" time (the time for retaliation), and reduces susceptibility to influence from course altering winds. For the low drag vehicle, the price for quick penetration of the earth's atmosphere is susceptibility to extreme heating.

As mentioned earlier, the dissipation of high kinetic energy during re-entry is in the form of heat. The re-entry vehicle, as it passes through the atmosphere, is preceded by a shock wave. This is caused by the molecules of air, which strike the forward surface of the vehicle, compressing (stagnating) them and
causing a rise in pressure which in turn causes a rise in temperature. Most of the resulting heat remains in the atmosphere with only a small percentage being absorbed by the vehicle. The shock layer heat that must be absorbed by the vehicle is transferred to it by convection. The resulting temperature at the most forward portion of the vehicle, the stagnation point, can exceed 8000 K. This shock wave extends laterally for a considerable distance beyond the vehicle, and results in a wake of hot air behind the vehicle that may contain as much as 99 percent of the heat. Shock waves precede blunt shapes more than streamlined shapes which result in a lower percentage of the total heat transferred to the vehicle. Unless some means of protection from the extreme temperatures caused by aerodynamic heating are offered, serious damage, if not total destruction, can result.

Flight tests were conducted in the early 1950's at Cape Canaveral (now Cape Kennedy) to solve the problems of re-entry. Early technology leaned toward the heat sink method of protecting craft from re-entry temperatures. The heat sink was essentially a metal shield which protected the payload by absorbing most
of the heat and radiating it back into the atmosphere. A main disadvantage of the heat sink was its weight. Adequate protection required a massive slab of aerodynamically shaped metal to soak up the high re-entry temperatures without transmitting them to the payload. The alternative plan was for re-entry cooling by ablation, a process by which heat is discarded in the deterioration of the heat shield. Fragments of the ablator flake off or burn away, carrying the heat with them. The first inflight demonstration of an ablative heat shield came on August 7, 1957 when a Jupiter C lofted a scale model Jupiter nose cone 600 miles high and 1200 miles downrange from Cape Canaveral. The recovered cone was shown to the nation on television three months later by President Dwight Eisenhower, who announced that the problem of re-entry had been solved.

Besides the ablation and heat sink method of heat protection, there is the radiator, radiation, liquid metal, magnetohydrodynamic and transpiration methods.

The radiator method is a regenerative cooling method in which a liquid is circulated through a jacket or series of tubes beneath the surface to be cooled and
then passed through a radiator or heat exchanger. Disadvantages to this method include excessive weight, complexity, low reliability and difficulty in maintaining.

The radiation method utilizes radiation cooling for heat dissipation along the re-entry body's surface. As the surface is heated by the airstream, some of the heat is radiated to space until an equilibrium temperature is reached. This method by itself does not have the capacity for handling high heat transfer rates unless extremely high surface temperatures are reached, then considerable insulation is required to reduce the heat that can flow to the interior of the re-entry body.

The liquid metal method of cooling is similar in principle to that of the radiator method. The main difference is that the use of certain liquid metals allow for a higher heat absorbing capacity per pound than is possible with nonmetallic liquids. It also allows for a greater range in temperature that may be handled. There are many problems involved in the use of such a system, some of which are excessive weight due to the liquid metal involved, complexity from pumping and piping, corrosive nature of dissimilar
metals and reliability.

The magnetohydrodynamic method utilizes the fact that during re-entry of a ballistic missile or space vehicle the resulting high temperatures cause the air immediately surrounding the re-entry body to become partially ionized. This ionized air is an electrical conductor. The principle of cooling by affecting the hydrodynamic flow of these ionized particles of air by the superposition of a magnetic field is called magnetohydrodynamic cooling. The main practical difficulties are the weight of the necessary electrical equipment and the cooling of some parts of this equipment.

The transpiration method of cooling consists of a system for forcing a fluid out through pores in the surface material. These pores can be produced mechanically or a porous material may be used. As the fluid passes through the pores the material is cooled, then as the fluid flows out over the surface it forms an insulating layer between the surface and the heat source. If the fluid vaporizes as it passes through the pores, additional heat is absorbed.

Important characteristics for this type of cooling are
1) a fluid with a high heat capacity and 2) a low heat transfer coefficient between the fluid and the surface of the re-entry body. Water used as a fluid in a transpiration system will absorb about five times as much heat as a solid copper re-entry body. Using water has a dual effect. As heat is absorbed and the water turns to steam, the layer of steam over the surface further insulates the surface against additional heating. The disadvantages of the transpiration method of heat protection are the high gross weight of the fluid to be carried and the weight of the delivery components that are necessary for the system to operate.

A variation of the transpiration cooling method is currently under investigation for use on re-entry vehicles and advanced terminal interceptors which essentially replaces the cumbersome water system \[1,2\]. If successful, it will alleviate the following problems associated with the nose tips of re-entry vehicles and advanced terminal interceptors: 1) the ability to withstand the extreme thermal environments generated by the heat transfer from the hot boundary layer in the high pressure area behind the bow shock, 2) the
requirements for a minimum shape change when subjected to
these enormous heat transfer rates both alone and in
combination with impact from rain, ice or dust particles,
3) enough strength to sustain the structural loads that arise from the asymmetrical pressure distributions induced by angles of attack up to 30 degrees, 4) the weight and volume consumed in the nose section of the vehicle by the coolant system necessary to assure nose tip survival with minimum recession (erosion of nose tip material) and 5) the complexity of the storage and ejection systems that deliver reliably the required flow of coolant to the nose tip in the high axial and possibly high lateral g environment of advanced re-entry vehicles.

The concept under study is referred to as GASJET Nose Tip and it basically replaces the complexity of the liquid delivery system with a warm gas system used in conjunction with a refractory metal nose tip. In this system shown in Figure 2, combustion products at 1600 to 2200 °F, generated from a solid propellant grain, are directly blown through a refractory tip thus eliminating expulsion systems and associated plumbing components. A refractory tip operating at 4000 to 5000 °F can be cooled by 1600 to 2200 °F gas.
Preliminary analysis for two typical re-entry vehicle trajectories show that significant savings in volume (50 percent), weight (20 percent) and complexity may be achieved over existing and projected water cooled nose tip systems [3]. In order to minimize the coolant flow rate and thus the over-all vehicle weight, it is desirable to operate the tip at the highest surface temperatures consistent with structural and oxidation limits.

Refractory metals were chosen for the nose tip structure because this group of metals offers high melting points that can be further enhanced with proper alloying and a high resistance to mechanical erosion. The principal refractory metals are shown in Table 1.

Widest experience has been gained with molybdenum alloys which show tensile strengths as high as 70,000 psi at 2000°F. The strongest molybdenum-matrix alloy tested at 2500°F contains small additions of titanium, zirconium and carbon. A columbium alloy with tungsten and tantalum additions displayed 30,000 psi strength at 2400°F, and a molybdenum alloy with a 25 percent tungsten addition tested at 73,500 psi at 2400°F.
Candidate materials for GASJET Nose Tip include tungsten-hafnium carbide (WHfc), 2% thoriated tungsten (W2ThO2) and tantalum-10% tungsten (Ta10W). Representative physical properties of these materials are summarized in Table 2.
II. TECHNICAL DISCUSSION

Even with the most advanced materials such as carbon/carbon, nose tip recession as much as several inches can occur at the velocities and altitudes of interest with sharp nose designs employing a passive nose tip cooling system. High recessions can even be expected when traversing rain or dust regions. This nose tip recession seriously affects the impact accuracy and control of a re-entry vehicle [4, 5]. If GASJET Nose Tip is to be considered a success, the use of the warm gas system in conjunction with a refractory tip should make possible zero or near zero recession of a nose tip in a thermal and/or particle environment.

Since re-entry vehicles and advanced terminal interceptors are typically non-recoverable, some means of measuring nose tip recession during test flights is necessary. The more conventional techniques for measuring this recession such as implanted radioactive sources, acoustic reflection, neutron backscatter,
electrical resistance, and electrical breakwires are fundamentally incompatible with the GASJET Nose Tip construction. A technique has been devised which permits stagnation point erosion to be determined from measured internal pressures. This approach works for a double choked flow passage leading up to the tip.

If such an approach is to be employed on an actual re-entry vehicle, the governing flow equations should first be developed, followed by verification testing of these flow equations and finally, a decision as to the feasibility/reliability based on reduced data from verification testing of the proposed recession measuring scheme.

Referring to Figure 3, the governing flow equations are developed under the following assumptions:

- $A_{THROAT} = \text{constant}$
- fluid temperature at tip and throat equal
- fluid is perfect gas
- isentropic flow
- double choked flow ($M_{TIP} = M_{THROAT} = 1$)
- $C_{D_{TIP}} = C_{D_{THROAT}}$
Flow through the tip can be written as:

\[
\dot{M}_{\text{tip}} = C_D^1 \frac{A P_T}{\sqrt{T_T}} \left[ \frac{\gamma g_c \left( \frac{2}{\gamma+1} \right) \delta+1}{R} \right]^{1/2}
\]

and flow through the throat can be written as:

\[
\dot{M}_{\text{throat}} = C_D^1 \frac{A P_T}{\sqrt{T_T}} \left[ \frac{\gamma g_c \left( \frac{2}{\gamma+1} \right) \delta+1}{R} \right]^{1/2}
\]

Equating II.1 and II.2 using the continuity equation, noting that \( P_c = P_{\text{throat}} \) and equating

\[
K = C_D^1 \left[ \frac{\gamma g_c \left( \frac{2}{\gamma+1} \right) \delta+1}{R} \right]^{1/2}
\]

\[
\frac{K A P_T}{\sqrt{T_T}} \left( \frac{A P_T}{\sqrt{T_T}} \right) = \frac{K A}{\sqrt{T_T}} P_c
\]

Assuming the discharge coefficient of the tip and the throat to be equal and double choked flow, equation II.3 reduces to:

\[
\frac{A P_T}{\sqrt{T_T}} = \frac{A}{\sqrt{T_T}} P_c
\]
If the flow rate of the coolant through the blast tube from the warm gas system is sufficiently small, the dynamic pressure can be assumed to be approximately zero making the static pressure measured in the blast tube the total pressure of the tip. Assuming also that the loss in heat between the throat and tip is small, the total temperature of the gas in the tip and throat are equal and the area of the throat is constant, equation II.4 reduces to:

$$A_{\text{TIP}} P_{BT} = A_{\text{THROAT}} P_c \quad \text{or} \quad P_{BT}/P_c = A_{\text{THROAT}}/A_{\text{TIP}} \quad \text{II.5}$$

Normalizing equation II.5:

$$\frac{P_{BT}/P_c}{(P_{BT}/P_c)_{\text{initial}}} = \frac{A_{\text{TIP}}}{A_{\text{TIP}}\text{ initial}}$$

where

$$A_{\text{TIP}} = \frac{\pi}{4} \left( \frac{D}{\text{TIP initial}} \right)^2$$
\[ A_{\text{TIP}} = \frac{\pi}{4} (D_{\text{TIP}_\text{initial}} + 2r \tan \theta)^2 \]

\[ = \frac{\pi}{4} (D_{\text{TIP}_\text{initial}}^2 + 4r \tan \theta D_{\text{TIP}_\text{initial}} + 4r^2 \tan^2 \theta) \]

Substituting values for \( A_{\text{TIP}} \) and \( A_{\text{TIP}_\text{initial}} \) into II.6:

\[
\frac{P}{P_{c}} = \frac{(P / P_{c})_{\text{initial}}}{\frac{\pi}{4} (D_{\text{TIP}_\text{initial}}^2 + 4r \tan \theta D_{\text{TIP}_\text{initial}} + 4r^2 \tan^2 \theta)}
\]

Dividing the right hand side by \( (D_{\text{TIP}_\text{initial}})^2 \):

\[
\frac{P}{P_{c}} = \frac{1}{\frac{1}{\frac{4r \tan \theta}{D_{\text{TIP}_\text{initial}}} + \frac{4r^2 \tan^2 \theta}{(D_{\text{TIP}_\text{initial}})^2}}} = \frac{1}{\frac{4r \tan \theta}{D_{\text{TIP}_\text{initial}}} + \frac{4r^2 \tan^2 \theta}{(D_{\text{TIP}_\text{initial}})^2}}
\]

II.7
Therefore, by knowing the initial flow diameter of the tip and the convergence half-angle of the flow passage of the tip, the measured values of blast tube pressure and gas generator chamber pressure will allow for the determination of the nose tip recession $r$.

Figure 4 shows a plot of $\frac{P_{BT}}{P_{c}}$ vs $\frac{(P_{BT})_{c \, initial}}{P_{c \, initial}}$.

This figure shows a sensitivity of blast tube pressure to recession and the convergence half-angle of the tip flow passage. As can be seen in the figure, as recession occurs, the flow area increases causing a drop in blast tube pressure.

With the governing flow equations developed, testing is needed to verify the scheme for measuring nose tip recession over a range of external hypersonic flow conditions. For these tests the Mach 6 and Mach 10 hypersonic wind tunnels at the NASA Langley Research Center were used. Tests were conducted using an undistorted tip to simulate no recession and three fore-shortened tips to simulate nose tip recession (see Figure 5). As in flight, pressures were measured in
the blast tube and the simulated gas generator chamber in order to substantiate and calibrate the sensitivity of internal pressures to recession of the tip. Static pressure at the tip exit ($P_{II}$) was measured during the wind tunnel testing to verify sonic flow at the tip. The objectives of these tests were to: 1) verify the scheme for measuring nose tip recession and 2) determine the range of mass flow ratios (mass flow of coolant gas to mass flow of free stream) required to sustain double choked flow.

The wind tunnel pressure models were full scale representations of the GASJET Nose Tip. They were mounted to a 6.3° half-angle cone simulating the forward section of a typical re-entry vehicle. The coolant flow was simulated by room temperature air introduced at the station corresponding to the gas generator location. The flow passage geometry from the throat to the tapered exit at the tip on the models were identical to those on an actual vehicle.
III. RESULTS

The wind tunnel testing performed at Langley verified the use of measuring internal pressures for determining nose tip recession. The range of mass flow ratios required to sustain choked flow at the tip was determined from internal pressure measurements made at the tip and in the blast tube.

Referring to equation II.7, using the known parameters \( r, D_{\text{tip}}, \theta \) from the foreshortened nose tips, the ratio \( \frac{P_{\text{BT}}/P_c}{(P_{\text{BT}}/P_c)_{\text{initial}}} \) can be calculated. Figure 6 shows excellent agreement between calculated and measured values of this ratio for the three foreshortened tips tested.

A calibration of the undistorted tip discharging into quiescent air was made in both the Mach 6 and Mach 10 wind tunnel. These results are given in Figures 7 through 9. When discharged into a near vacuum, both the tip and the simulated gas generator orifice became sonic almost instantaneously. Subsonic flow exists up to a flow of approximately 0.008 lb/sec
for the tip discharging into the atmosphere as shown in Figures 7 and 8. The supply pressure variation, Figure 9, shows the upstream orifice is sonic at a flow of approximately 0.004 lb/sec for a room pressure discharge. The calibration seems to be reasonable since the gas generator orifice area is half the area of the tip flow passage.

Operation of the undistorted tip model in the Mach 6 and Mach 10 free stream resulted in internal pressure ratios as shown in Figures 10 through 14. As mass flow ratio increased, the measured static pressure at the exit of the tip decreased to a ratio of 0.65 when sonic flow was established at the tip. Taking the minimum exit pressure ratio as 0.65 gives a matching exit Mach number of 0.8 observed in the ambient discharge calibration. If model measurements could have been located at the precise point of sonic flow, it would then be expected a minimum pressure ratio of 0.53 would be reached.

Internal pressure data for the three fore-shortened tips are shown in Figures 15 through 29. The mass flow ratio at which the recessed tip's flow becomes sonic is constant with each Mach number.
IV. CONCLUSIONS

With excellent agreement between calculated and measured values for the proposed nose tip recession measuring scheme, it can be concluded that the proposed method can be used confidently on a hypersonic test vehicle. The determination of the required chamber pressure to stagnation pressure for double choked flow in a hypersonic flow field verifies the possibility of using a gas generator to provide the required flow rate of warm coolant gas.

However, one should be aware of possible sources of error when determining recession by this method. Possible sources of error are: 1) resolution in telemetry (during re-entry, various data, including the required internal pressure measurements will be telemetered), 2) tolerances on telemetry calibrations, 3) tolerances on pressure transducer calibration, 4) manufacturing tolerance, i.e., throat diameter and convergence angle, 5) thermal expansion/contraction, 6) non-symmetrical recession, 7) peening of tip to reduce flow area, and 8) leakage.
APPENDIX I

FIGURES
Figure 1. Effect of $W/C_D A$ and Altitude on Re-entry Vehicle
Figure 2. GASJET Nose Tip Configuration
Figure 3. GASJET Nose Tip Flow Passage
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Figure 5. Wind Tunnel Test Models
Figure 6. Sensitivity of Blast Tube Pressure to Recession ($\theta = 1.5^\circ$)
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Figure 8. Ambient Discharge Pressure in Blast Tube
Figure 9. Ambient Discharge Pressure at Supply
Figure 10. GASJET Tip Pressure Ratio vs Mass Flow Ratio
Figure 11. GASJET Tip Pressure Ratio vs Mass Flow Ratio

$\frac{P_{T_1}}{P_{T_2}}$

Internal Pressure Ratio

$M = 10.4$
Model A
Figure 12. Blast Tube Pressure Ratio vs Mass Flow Ratio
Figure 13. Blast Tube Pressure Ratio vs Mass Flow Ratio
Figure 14. Supply Pressure Ratio vs Mass Flow Ratio

M = 10.4
Model A
Figure 15. GASJET Tip Pressure Ratio vs Mass Flow Ratio
Figure 16. GASJET Tip Pressure Ratio vs Mass Flow Ratio

$P_{T1}/P_{T2}$
Internal Pressure Ratio

$M = 10.4$
Model B

Mass Flow Ratio ($\dot{m}$)
Figure 17. Blast Tube Pressure Ratio vs Mass Flow Ratio
Figure 18. Blast Tube Pressure Ratio vs Mass Flow Ratio
Figure 19. Supply Pressure Ratio vs Mass Flow Ratio

\[ \frac{P_{sup}}{P_{T_2}} \]

Supply Pressure Ratio

\[ M = 10.4 \]

Model B

Mass Flow Ratio (m)
Figure 20. GASJET Tip Pressure Ratio vs Mass Flow Ratio
Figure 21. GASJET Tip Pressure Ratio vs Mass Flow Ratio
Figure 22. Blast Tube Pressure Ratio vs Mass Flow Ratio
Figure 23. Blast Tube Pressure Ratio vs Mass Flow Ratio
Figure 24. Supply Pressure Ratio vs Mass Flow Ratio

\[ \frac{P_{\text{sup}}}{P_{T_2}} \]

Supply Pressure Ratio

\[ M = 10.4 \]

Model C
Figure 25. GASJET Tip Pressure Ratio vs Mass Flow Ratio
Figure 26. GASJET Tip Pressure Ratio vs Mass Flow Ratio
Figure 27. Blast Tube Pressure Ratio vs Mass Flow Ratio
Figure 28. Blast Tube Pressure Ratio vs Mass Flow Rate

\[ \frac{P_{BT}}{P_{T_2}} \]

Internal Pressure Ratio

\[ M = 10.4 \]

Model D
Figure 29. Supply Pressure Ratio vs Mass Flow Ratio
<table>
<thead>
<tr>
<th>METAL</th>
<th>ATOMIC WEIGHT</th>
<th>MELTING POINT (°F)</th>
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<tr>
<td>Columbium</td>
<td>92.91</td>
<td>3542</td>
</tr>
<tr>
<td>Hafnium</td>
<td>178.6</td>
<td>3092</td>
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<td>Tantalum</td>
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<td>Tungsten</td>
<td>183.9</td>
<td>6116</td>
</tr>
<tr>
<td>Zirconium</td>
<td>91.22</td>
<td>3452</td>
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**TABLE 1. PRINCIPAL REFRACTORY METALS**
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<thead>
<tr>
<th>Property</th>
<th>Ta10W</th>
<th>W2ThO₂</th>
<th>WHfC</th>
</tr>
</thead>
<tbody>
<tr>
<td>Melting Point Temperature (°F)</td>
<td>5495</td>
<td>6200</td>
<td>6200</td>
</tr>
<tr>
<td>Density (lbm/in³)</td>
<td>0.608</td>
<td>0.675</td>
<td>0.697</td>
</tr>
<tr>
<td>Thermal Conductivity (BTU-ft/sec ft⁻² R⁻¹)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>@ 2500 °F</td>
<td>0.011</td>
<td>0.024</td>
<td>0.024</td>
</tr>
<tr>
<td>@ 4000 °F</td>
<td>0.012</td>
<td>0.023</td>
<td>0.023</td>
</tr>
<tr>
<td>Brittle-to-Ductile Transition Temperature (°F)</td>
<td>-320</td>
<td>400 to 950</td>
<td>400 to 950</td>
</tr>
<tr>
<td>Ultimate Tensile Strength (KSI)</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>@ 80 °F</td>
<td>160</td>
<td>90</td>
<td>180</td>
</tr>
<tr>
<td>@ 4000 °F</td>
<td>5</td>
<td>10</td>
<td>30</td>
</tr>
<tr>
<td>@ 4500 °F</td>
<td>2.5</td>
<td>6</td>
<td>8</td>
</tr>
</tbody>
</table>

**TABLE 2. CANDIDATE REFRACTORY METAL PROPERTIES**
LIST OF REFERENCES


BIBLIOGRAPHY


