Hybrid Rocket Engines – Combustion Analysis – Independent Research

ENGR 594

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# Abstract

Understanding the current state of the rocket industry is essential to evaluate how human life will be transformed. The rocket engine market is currently dominated by solid rocket engines and liquid rocket engines. This thesis aims to understand the hybrid rocket engine better, incorporating the two designs. The project is to determine the benefits and deficiencies of the hybrid rocket engine compared to the other two options.

# Introduction

What is propulsion? In a broad sense, propulsion is the act of changing the motion of a body. Propulsion mechanisms provide a force that moves bodies initially at rest, changes a velocity, or overcomes drag forces when mass is propelled through a medium. Jet propulsion is a means of a reaction force bestowed to a device by the momentum of discharged matter. Rocket propulsion is a class of jet propulsion that produces thrust by ejecting stored matter, called the propellant.

Rockets are hazardous propulsion systems, and when a failure occurs, loss of life usually follows. For getting heavy objects to space, there is no alternative currently. The rocket engine excels above any other propulsion system when high thrust and high acceleration are needed to move an object. We need to generate a significant amount of velocity in order to orbit the earth or leave the earth's gravitational influence.

Characterizing the combustion chemistry of a rocket motor is critical—the understanding of how fuel and oxidizer react at specific temperatures. The two most common forms of chemical propellent include LREs (Liquid Rocket Engine) and SREs (Solid Rocket Engine). HREs (Hybrid rocket engines) are “amongst the most promising systems using chemical rocket propellant and possessing practically all merits of up-to-date LREs and SRMs are hybrid rocket engines running on solid fuel and liquid or gaseous oxidizer.” [1] HREs are an intermediate between LRE and SRM by controlling the flow of an oxidizer with an SRM, which gives the system the ability to control the combustion cycle and performance while having increased thrust, Isp, and mission capabilities (Δv).

Hybrid propellants will be chosen based on density, greatest thermal release, and lowest molecular weight, ensuring the design can meet the highest achievable delta-V. “Propellants based on polymer HC fuels (synthetic rubbers)

with liquid oxygen or concentrated hydrogen peroxide are most highly mastered and, at the same time, efficient enough. To raise power characteristics and stabilize combustion, metals (or metals hydrides) can be added to solid components. These may be aluminum, boron, lithium." [1] With our choice of propellant, we will need to consider an economical and sustainable option. The cost per pound to low-earth orbit is a factor for justification for the HRE.

(1)

(2)

The specific impulse is the total impulse per unit weight of propellant. Specific impulse is similar in concept to the miles per gallon parameter used with automobiles. A higher number translates to better performance. Specific impulse gives us a quick way to determine the thrust of a rocket if we know the weight flow rate through the nozzle. It is an indication of engine efficiency. Two different rocket engines have different values of specific impulse. The engine with a higher specific impulse value is more efficient as it produces more thrust for the same amount of propellant. Specific impulse gives us an easy way to "size" an engine during preliminary analysis. The result of our thermodynamic analysis is a particular value of a specific impulse. The rocket weight will define the required value of thrust. Dividing the thrust required by the specific impulse will tell us how much weight flow of propellants our engine must produce. This information determines the physical size of the engine.

Hybrid rocket engines have a significant advantage over solid rocket engines due to their ability to restart. Since the oxidizer can be controlled through a valve, the thrust can be controlled. Solid propellants pose a severe hazard during transportation due to their potential for catastrophic failure due to severe vibrations. Since the solid ' fuel' in a hybrid engine has very little or no oxidizers or other non-polymeric substances embedded in it, it is highly elastomeric and can stand much greater vibration. When injecting the fuel and oxidizer, improper timing can result in a failure mode with liquid rocket engines. The development of high thrust liquid rocket engines is beset with the severe problem of combustion instability due to high frequencies. Removal of instabilities is considered very time and money-consuming. The Saturn V (F-1 Engine) development to reduce high-frequency instabilities was very costly and time-consuming.

Investigation into the efficiencies of the oxidizer flow will be an area of interest for determining the optimal performance of the HRE. As the oxidizer flows over the boundary layer of the solid fuel, this will be the area of combustion and create the combustion boundary layer. “The calculation procedure

incorporates solving equations of the turbulent boundary layer for the binary chemically-reacting mixture.” [1] By understanding the interaction at the boundary layer, predictions can be made to quantify the efficiencies of the HRE design. Thus, experimental research can be performed to investigate the prediction models further. In this study, we will not focus on the details of the combustion boundary layer interactions but rely on computational modeling of the fuel/oxidizer combustion process.

This study will analyze a combination of solid, liquid, and hybrid rocket propellant options - "Paraffin fuel is selected due to its high regression rate. Previous experiments have shown that the regression rate of paraffin-based fuel is three to four times higher than classical hybrid fuels. Liquid nitrous oxide as oxidizer offers inherent advantages such as it is relatively cheap and available, non-toxic, self-pressurizing, and subcritical (storable) at room temperature" [2]. This study will use CEA software to produce results and compare the options to determine if paraffin wax meets any large-scale production needs.

Chart

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**Figure 1 – Reference HRE Combustion Boundary Layer [1]**

# Comparative Evaluation

The performance of a rocket can be computed by using the "Ideal Rocket Equation ."This ideal rocket equation is as follows:

(3)

The ideal rocket equations may be viewed as inserting into the vehicle a velocity increment where g is the acceleration due to gravity, is the structural weight, Wp is the propellant, and is the specific impulse, which is defined as the impulse per unit weight of the propellant and can also be viewed as the thrust per unit weight flow rate. This delta-V equation provides a measure of the capability of a vehicle to complete an orbital maneuver. For example, a launch from Cape Canaveral, FL, to low earth orbit may have a delta-V requirement of 7.8 km/s. A vehicle with a delta-V of 7.8 km/s or more would be able to complete this mission profile.

Understanding how to increase will be of our greatest interest for generating the most performance of our design. However, if we only increase specific impulse, we will be ignoring our density impulse (ρp). Large values of achieved imply less propellant to be carried, and large values of ρp imply less inert weight to carry the same propellant weight. In general, it is not possible to obtain propellants with large values of both and ρp

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**Figure 2 – Performance of different propellants**

Figure 2 shows the values of ρp, , ρp, and (O/F)opt for several different propellant combinations. (O/F)opt is the weight ratio of the oxidizer-to-fuel needed to reach the maximum specific impulse for the propellant. Figure 2 propellant combinations represent those that are available for manufacturing. From Figure 2, we can see that solid propellants generally have a lower specific impulse than liquid propellants. We can also observe that Figure 2 shows hybrid engines have a higher and lower (Opt)O/F. Thus, this would tell us that the hybrid engine design would be a better solution for our goal of high performance and efficiency.

|  |  |  |  |
| --- | --- | --- | --- |
| **Design Factors** | **Solid Rocket** | **Liquid rocket** | **Hybrid Rocket** |
|  | Good | Best | Good |
| **Pressure Dependence** | Dependent | Dependent | Not dependent |
| **Throttleability** | None | Full | Full |
| **Safety/Transportation** | Large Risk | Large Risk | Less Risk |
| **Cost** | Medium | High | Medium |
| **Fuel Research** | High | High | Low |
| **Thrust** | High | Medium | Theoretically High |

**Figure 3 – Characteristics of various propulsion systems**

Another topic of consideration is the logistics of handling and transportation of propellants. Evaluation of the different options will help determine the overall feasibility of costs. Hybrids appear distinctly superior to solids and marginally better compared to liquids. Solid propellants pose a severe hazard during transpiration due to severe vibrations' potential for catastrophic failure. One would then assume the same would be valid for hybrid rocket engines, but since the solid 'fuel' in a hybrid engine has very little to no oxidizers or other non-polymeric substances embedded in it, it is highly elastomeric and can stand much greater vibration. Another important point concerns the problem of high-frequency combustion instability. In recent times, however, the problem is significant; high-frequency instability (frequencies up to 5-6 kHz and amplitudes of about 2-10 % of mean pressure) appears to be present in the existing weapon systems (Karnesky & Colucci 1975). Liquid engines can be prone to similar problems when they experience high-frequency instability during their development. Significant engineering time and money are spent on engineering out the vibration concerns for the liquid engine design. Hybrid engines during development have not shown any of the similar vibration failure modes as the liquid and solid rocket engine design. We will next look at the mass flow rate of the different rocket designs.

(4)

(5)

(6)

Where Ab is the burning area, is the oxidizer flow rate, is the fuel flow rate, is the linear burning rate, and is the linear regression rate. Though the terms and both refer to the same combustion rate, applies to solid propellants where the deflagration occurs autonomously of the external oxidizer. In contrast, applies to hybrid propellants when the fuel regresses due to the heat flux from the diffusion flame or hot gases. The dependence of the terms and on parameters like pressure and mass flux is governed by mixing and chemical reactions. The more mass we can flow through the exhaust, the higher the specific impulse we generate and the higher delta-V we can achieve.

Since solid propellants have finely mixed fuel and oxidizers, the is vital for the chemical reaction rate. The chemical reaction depends strongly on pressure, as the location of the combustion zone will move with respect to the surface. Due to abnormalities in manufacturing and changes in volume, the burning rate of solid propellants depends strongly on pressure. This dependence empirically represents most propellants in the acceptable pressure range as , where and n are constants. Typical values of and n can be referred to in Figure 3. The constants can be altered by introducing small amounts of modifiers to the chemical makeup in the solid propellants. The pre-mixed state of fuel and oxidizer causes the propellant to burn along all the exposed surfaces of the propellant. If any minor manufacturing defects (such as a crack) in the propellant are present, the mass production rate will become much more prominent than designed. Thus, this could lead to a pressure vs. time curve that the vehicle was not designed to handle. The result could lead to the events of a complete failure.

The combustion process of the liquid rocket engine can be more complex than that of the solid rocket engine. The combustion process of a liquid engine injects the propellants into the combustion chamber. The fuel and oxidizer react intensely if the propellants are hypergolic or atomize to help quick vaporization. Chemical reactions occur in the gas phase, leading to the generation of hot gases after the vaporized components get mixed. The liquid rocket engine processes involve mixing, vaporization, and chemical reaction; we will see that vaporization is taken to be the rate-limiting mechanism for determining the overall parameters of the engine design. One of the significant variables to consider in the combustion chamber is droplet spray vaporization in the hot gaseous environment. This process is diffusion-controlled; therefore, the effects of kinetics will be secondary. When injecting the fuel and oxidizer, improper timing can result in a failure mode with liquid rocket engines. The development of high thrust liquid rocket engines is beset with the severe problem of combustion instability due to high frequencies. Although the present state-of-the-art permits trying some solutions, removing instabilities is considered very time and money consuming.

The hybrid rocket engine process in a hybrid rocket combustion chamber creates the combustion in the boundary layer close to the solid fuel surface (figure 1). The initial combustion phase consists of the processes of entrenchment of the liquid oxidizer on the fuel surface and its vaporization. Since the process is diffusion-dominated, chemical kinetics and pressure have a relatively more minor impact on the engine's performance. This process is the limiting factor in a solid rocket engine design. Hybrid rocket fuels often drive multi-port fuel grains. Multi-port fuel grains have poor volumetric efficiency and structural deficiencies. Exploring some of the features in Figure 3, we can determine that the O/F performance of a hybrid engine is not constant throughout flight. Using the equation in [3], we can express O/F in terms of the inner diameter of the cylindrical fuel block burning from inside outwards as

(7)

where d is equal to the inner diameter of the port and L is the length of the fuel. If n=0.5, as in the ease of laminar flow, O/F is a constant and does not change during the combustion process. If n = 0.8, as in the case of turbulent flow, the value of O/F increases during the firing, showing that the products become oxidizer-rich during the combustion process. Since the O/F is changing during the combustion process, the specific impulse of the system is also changing. One of the potential options is to fix the initial operating point on a slightly fuel-rich side so that when the operating point moves to the oxidizer-rich side, the specific impulse does not vary by more than 1-2%. Another option to maintain a constant O/F level is to use two oxidizer injection points, one near the head end and the other near the exhaust end. At the beginning of the combustion process, the burning is fuel-rich, and exhaust end injection is used to optimize it. At the end of the combustion process, the end of the fuel block tends towards oxidizer-richness, so exhaust end injection is reduced to maintain the same O/F level. The second feature of the hybrid rocket engine design is the low explosion hazard during storage, transportation, and combustion. The hybrid has the solid fuel and liquid oxidizers separately stored, which is different from the solid rocket engine design. Since the two are separate, it is nearly impossible for an accidental agitation that could ignite the engine. In an accidental agitation, the solid rocket engine can burn by itself, whereas the fuel in the hybrid rocket must receive an oxidizer for its combustion. The fire hazard of the hybrid is more minor than that of liquid rocket engines because, in the event of an explosion, the liquids of a liquid rocket are free to flow and get mixed up. The hybrid rocket fuel and oxidizer have greater miscibility to large-scale mixing since the fuel is in the form of a solid. Another comparison to solid is that any defects from manufacturing, such as cracks, could lead to catastrophic failure, which is not true for hybrids. We will notice that the regression of the fuel occurs under the action of a heat flux from the combustion front. Thus, any fuel area that experiences less heat will regress less comparatively. The third feature of the hybrid rocket engine concerns the sensitivity of the regression rate to the compounds of fuel. A solid rocket engine cannot perform as designed when the compounds or manufacturing are not made to the specifications of the chemical makeup. If this is off, it can lead to total failure. This is not the same for the hybrid rocket engine design. The regression rate is negligibly dependent on the nature of fuels, even when they are different.

We can summarize the importance of the O/F ratio as we compare it to the temperature inside the chamber. As the temperature reaches the peak value, we can be closest to our stoichiometric condition. With this condition, we can compare the specific impulse and determine if we have the best performance at our most efficient point.

It is hard to understand why hybrid engines have not been used in any large-scale launch vehicle system despite presenting the advantages discussed above. A study of early literature reveals that the answer lies more in history than in technology. Promising developmental work on hybrid engines was completed in 1963-1964. The USA had developed solid/liquid engines with thrust levels far exceeding that of hybrid engines (Figure 4)

Table

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**Figure 4 – Engines available in 1963**

These time periods were a strategic competition between the USA and the USSR to build larger and larger thrust engines for military applications. This conflict led to increased investments in current technological systems that were difficult to obtain sufficient investments into new propulsion systems. One could determine that the future of hybrid rockets, as with any other propulsion system, is difficult to predict because political factors interfere with technical factors. Significant investments have already been spent developing liquid and solid propellant rockets. It is difficult to imagine a similar effort for hybrid propulsion systems now. Our collective understanding of an engineered system is closely related to the amount of money spent. We could conclude that if the investments were made for hybrids instead of solid or rocket, we would have the same understanding of hybrids as we do now of solid and liquid rocket engine designs. We would then be asking whether it was of interest to develop solid or liquid motors. However, this is not a valid reason to ignore hybrid rockets now. The potential performance and multiple outlined benefits argue persuasively for increasing investment and a continuation of the current program of research and development.

# **Analysis**

Hybrid rocket engines are designed with a low regression rate. It is helpful to analyze some of the different configurations to help minimize the impact of the low regression rate on the rocket engine's performance. A mixture of hydrocarbons, wood, wax, and other fuels was used to study the hybrid rocket engine design. It is to be noted that acryl and polyethylene are still used for laboratory experiments. More common is using Hydroxyl-terminated polybutadiene (HTPB) as it is safe, easy to obtain, and a mixture of resin and curing agent. This agent can be further mixed with other chemicals to increase the mechanical strength of the load, density, and other powders to alter the performance. Rocket fuels using HTPB/LOX as a solid fuel obtain a specific impulse greater than solid rocket engines but lower than liquid fuel engines such as LO2/LH2.

HTPB/LOX ensures the performance of the rocket engine will be close to the LOX/kerosene combination. However, reviewing HTPB/LOX further, we can determine it is an expensive fuel, expensive on a large scale, and has an environmental impact.

**Table

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**Figure 5 – Merit Performance [9]**

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
|  | **LNG** | **Paraffin** | **RP-1** | **Solid (HTPB/AP/ µAl)** |
| ***I*s, vacuum[s]** | 369 | 340 | 358 | 290 |
| **Density [kg/m3]** | 820 | 900 | 1030 | 1770 |
| **Cost per unit mass [€/kg]** | .5 | 0.8 | 3.3 | 5 |
| **Gravity acceleration [m/s2]** | 9.81 | | | | |

**Table 1 – Reference Data for Different Fuels [9]**

“Is, vacuum, gravity acceleration, and density are used to calculate volumetric specific impulse as [kN s/m3]. Density is used to calculate cost per unit volume from cost per unit mass. Considering a merit parameter [kN·s/€], paraffin fuels give a value of 4.17, which is approximately four times RP-1 (1.06) and eight times HTPB/AP/µAl solid fuels (0.57). With a value of 7.24, LNG has the highest merit, but this calculation does not consider the significant additional system and management costs related to cryogenic liquid fuels.” [9] A further investigation into Paraffin is needed to compare more of the performance numbers of the fuel. This investigation proves that the Paraffin outperforms solid fuels, which are commonly used for low earth orbit satellite deployment, and is cheaper costs. We can see this evaluation further in Figure 1, which depicts the merit performance as a performance per dollar chart.

Using the NASA CEA model, the analysis can be complete to compare the solid, liquid, and hybrid rocket performance parameters. Using the NASA CEA tool, the pressure will be kept as a constant of 8bar of pressure. The supersonic area ratio will be kept at 8. Fuel and oxidizer will be held at 100% weight during the combustion process. The option that will be changing is the O/F ratio better to understand the best performance for the different engine types. The O/F ratio will be changed from 1 to 8 in increments of 8.

**Figure 6 – LH/LOX O/F Performance**

**Figure 7 – Paraffin Wax N2O O/F Performance**

**Figure 8 – ALNH4CLO4(I) O/F Performance**

|  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- |
| **Liquid (LH/LOX)** | | | | | |
| O/f | Isp, M/SEC | CSTAR, M/SEC | Ivac, M/SEC | T (K) CHAMBER | Cp Chamber |
| 1 | 3241.5 | 2092 | 3420 | 977.49 | 7.8107 |
| 2 | 3682.7 | 2363.1 | 3907.3 | 1797.25 | 6.2719 |
| 3 | 3800.5 | 2424.5 | 4064.1 | 2430.88 | 6.1171 |
| 4 | 3974.4 | 2405.1 | 4091.5 | 2853.8 | 7.6031 |
| 5 | 3719.1 | 2340.2 | 4046.8 | 3102.15 | 10.0061 |
| 6 | 3600.8 | 2258 | 3954.4 | 3233.14 | 12.6107 |
| 7 | 3465.8 | 2173.1 | 3829.8 | 3287.29 | 13.9645 |
| 8 | 3338.1 | 2093.1 | 3693.6 | 3295.74 | 13.5129 |
| **Hybrid (Paraffin/NOX)** | | | | | |
| O/f | Isp, M/SEC | CSTAR, M/SEC | Ivac, M/SEC | T (K) CHAMBER | Cp Chamber |
| 1 | 1536.3 | 964.9 | 1692.8 | 1027.92 | 8.1621 |
| 2 | 1682.1 | 1057.5 | 1854 | 1161.32 | 5.6069 |
| 3 | 1757.8 | 1106.3 | 1933.1 | 1287.79 | 2.8174 |
| 4 | 1883.5 | 1211.9 | 2046 | 1736.30 | 1.6684 |
| 5 | 2048.8 | 1309.5 | 2196.4 | 2174.59 | 1.6805 |
| 6 | 2144 | 1367.9 | 2302.8 | 2494.93 | 1.8008 |
| 7 | 2206 | 1406.2 | 2373.9 | 2704.90 | 2.2288 |
| 8 | 2242 | 1422.8 | 2419.7 | 2784.43 | 3.1697 |
| 9 | 2243.7 | 1410.9 | 2436.5 | 2771.53 | 3.458 |
| 10 | 2194.5 | 1385.1 | 2375.6 | 2717.07 | 3.1421 |
| 11 | 2136.4 | 1354.2 | 2305.5 | 2641.70 | 2.6917 |
| 12 | 2079.5 | 1321.6 | 2239.2 | 2555.10 | 2.3403 |
| 13 | 2025.8 | 1289.3 | 2177.8 | 2463.36 | 2.0198 |
| 14 | 1975.9 | 1258.6 | 2121.3 | 2371.02 | 1.8245 |
| 15 | 1926.6 | 1229.9 | 2069.2 | 2281.25 | 1.6929 |
| **Solid (AL/NH4CLO4)** | | | | | |
| O/f | Isp, M/SEC | CSTAR, M/SEC | Ivac, M/SEC | T (K) CHAMBER | Cp Chamber |
| 1 | 2194.4 | 1372 | 2442.5 | 3980.11 | 10.4503 |
| 2 | 2406.6 | 1504.7 | 2679.1 | 3974.32 | 20.2263 |
| 3 | 2455.4 | 1537.0 | 2726.7 | 3853.81 | 14.2620 |
| 4 | 2463.1 | 1543.2 | 2731.7 | 3730.45 | 10.7150 |
| 5 | 2459.7 | 1541.7 | 2725.6 | 3625.13 | 8.8392 |
| 6 | 2452.8 | 1537.9 | 2715.9 | 3537.92 | 7.7051 |
| 7 | 2444.9 | 1533.2 | 2704.9 | 3465.46 | 6.9416 |
| 8 | 2436.9 | 1528.5 | 2693.7 | 3404.56 | 6.3890 |

**Table 2 – O/F Performance**

**Figure 9 – ISP Performance**

**Figure 9 – CSTAR Performance**

The data shows that the performance numbers for the LH/LOX option are showing better performance for the ISP, CSTAR, and IVAC. However, since this engine is very efficient, we know that it does not maintain the same thrust levels as the solid rocket. We can note this by real-world examples of the Space Launch System (SLS). Noting from the solid rocket is that it performs similarly on paper to the paraffin wax model. We can note that the two engines are similar in performance, with an O/F ratio of 8 for paraffin wax and 3 for the ALNH4CLO4(I) model. The two designs have similar Isp (Ivac) and CSTAR performance numbers.

While we can note that the specific impulse is higher for the LH/LOX engine design, we note from Figure 2 that the energy density is also a factor. When we evaluate the energy density multiplied by the specific impulse, we can see that the hybrid rocket engine's performance surpasses the liquid rocket design but less than the solid rocket design. We can note that we will see similar thrusts to the solid rocket engine with more safety and the ability to throttle the engines.

The other conclusion from this data is that the paraffin wax model did not level off like the other two designs. We can determine from the paraffin wax model that the performance increases as the O/F ratio increases. This means that while the other two designs are best performing in a range of 2-4 O/F ratio, the paraffin wax model needs further analysis to determine the best O/F ratio. We can note that the range between the numbers as the O/F was decreasing, which would point us to being close to the peak performance of the design.

# **Conclusion**

A review of hybrid propulsion systems has been presented from the point of view of research and development. A review of the CEA model indicates that the hybrid rocket engine with a relatively common item can perform as well as a solid rocket engine design. We can note this because the paraffin wax option had a similar ISP to the solid rocket engine design. Based on our research, we determined that large-impulse systems are technically feasible and economically viable for the solid rocket engine design. Further research and development are needed to investigate the hybrid design further to determine what fuels will be of the most performance and scalability for manufacture. The reasons for overlooking the development of the hybrid propulsion system have been brought out to be more historical than technical.

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# **Appendix**

|  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- |
| \*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\* |  |  |  | |  |
|  |  |  |  | |  |
| NASA-GLENN CHEMICAL EQUILIBRIUM PROGRAM CEA2, FEBRUARY 5, 2004 |  |  |  | |  |
| BY BONNIE MCBRIDE AND SANFORD GORDON |  |  |  | |  |
| REFS: NASA RP-1311, PART I, 1994 AND NASA RP-1311, PART II, 1996 |  |  |  | |  |
|  |  |  |  | |  |
| \*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\* |  |  |  | |  |
|  |  |  |  | |  |
|  |  |  |  | |  |
|  |  |  |  | |  |
|  |  |  |  | |  |
| ### CEA analysis performed on Sat 23-Apr-2022 16:14:04 |  |  |  | |  |
|  |  |  |  | |  |
| # Problem Type: "Rocket" (Infinite Area Combustor) |  |  |  | |  |
|  |  |  |  | |  |
| prob case=\_\_\_\_\_\_\_\_\_\_\_\_\_\_\_9924 ro equilibrium |  |  |  | |  |
|  |  |  |  | |  |
| # Pressure (1 value): |  |  |  | |  |
| p,bar= 8 |  |  |  | |  |
| # Supersonic Area Ratio (1 value): |  |  |  | |  |
| supar= 8 |  |  |  | |  |
|  |  |  |  | |  |
| # Oxidizer/Fuel Wt. ratio (8 values): |  |  |  | |  |
| o/f= 1, 2, 3, 4, 5, 6, 7, 8 |  |  |  | |  |
|  |  |  |  | |  |
| # You selected the following fuels and oxidizers: |  |  |  | |  |
| reac |  |  |  | |  |
| fuel H2(L) wt%=100.0000 |  |  |  | |  |
| oxid O2(L) wt%=100.0000 |  |  |  | |  |
|  |  |  |  | |  |
| # You selected these options for output: |  |  |  | |  |
| # short version of output |  |  |  | |  |
| output short |  |  |  | |  |
| # Proportions of any products will be expressed as Mass Fractions. |  |  |  | |  |
| output massf |  |  |  | |  |
| # Heat will be expressed as siunits |  |  |  | |  |
| output siunits |  |  |  | |  |
|  |  |  |  | |  |
| # Input prepared by this script:/var/www/sites/cearun.grc.nasa.gov/cgi-bin/CEARU |  |  |  | |  |
| N/prepareInputFile.cgi |  |  |  | |  |
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| ### IMPORTANT: The following line is the end of your CEA input file! |  |  |  | |  |
| end |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
|  |  |  |  | |  |
| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 1.00000 %FUEL= 50.000000 R,EQ.RATIO= 7.936683 PHI,EQ.RATIO= 7.936683 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.8745 93.669 |  |  |  | |  |
| P, BAR 8.0000 4.2677 0.08541 |  |  |  | |  |
| T, K 977.49 826.58 281.10 |  |  |  | |  |
| RHO, KG/CU M 3.9686-1 2.5036-1 1.4747-2 |  |  |  | |  |
| H, KJ/KG -2438.06 -3604.54 -7691.60 |  |  |  | |  |
| U, KJ/KG -4453.89 -5309.16 -8270.77 |  |  |  | |  |
| G, KJ/KG -41166.4 -36353.9 -18828.6 |  |  |  | |  |
| S, KJ/(KG)(K) 39.6202 39.6202 39.6202 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 4.032 4.032 4.035 |  |  |  | |  |
| MW, MOL WT 4.032 4.032 4.032 |  |  |  | |  |
| (dLV/dLP)t -1.00000 -1.00000 -1.14313 |  |  |  | |  |
| (dLV/dLT)p 1.0000 1.0000 3.7385 |  |  |  | |  |
| Cp, KJ/(KG)(K) 7.8107 7.6564 115.2339 |  |  |  | |  |
| GAMMAs 1.3587 1.3686 1.1195 |  |  |  | |  |
| SON VEL,M/SEC 1655.0 1527.4 805.2 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 4.026 |  |  |  | |  |
|  |  |  |  | |  |
| PERFORMANCE PARAMETERS |  |  |  | |  |
|  |  |  |  | |  |
| Ae/At 1.0000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2092.0 2092.0 |  |  |  | |  |
| CF 0.7301 1.5494 |  |  |  | |  |
| Ivac, M/SEC 2643.4 3420.1 |  |  |  | |  |
| Isp, M/SEC 1527.4 3241.5 |  |  |  | |  |
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|  |  |  |  | |  |
| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H2 0.43700 0.43700 0.43700 |  |  |  | |  |
| H2O 0.56300 0.56300 0.55897 |  |  |  | |  |
| H2O(L) 0.00000 0.00000 0.00403 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
|  |  |  |  | |  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
|  |  |  |  | |  |
| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 2.00000 %FUEL= 33.333333 R,EQ.RATIO= 3.968341 PHI,EQ.RATIO= 3.968341 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.8301 84.200 |  |  |  | |  |
| P, BAR 8.0000 4.3714 0.09501 |  |  |  | |  |
| T, K 1797.25 1570.14 601.42 |  |  |  | |  |
| RHO, KG/CU M 3.2375-1 2.0250-1 1.1491-2 |  |  |  | |  |
| H, KJ/KG -1760.57 -3158.02 -8541.86 |  |  |  | |  |
| U, KJ/KG -4231.63 -5316.72 -9368.70 |  |  |  | |  |
| G, KJ/KG -58531.6 -52755.3 -27539.2 |  |  |  | |  |
| S, KJ/(KG)(K) 31.5878 31.5878 31.5878 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 6.047 6.048 6.048 |  |  |  | |  |
| MW, MOL WT 6.047 6.048 6.048 |  |  |  | |  |
| (dLV/dLP)t -1.00003 -1.00000 -1.00000 |  |  |  | |  |
| (dLV/dLT)p 1.0008 1.0001 1.0000 |  |  |  | |  |
| Cp, KJ/(KG)(K) 6.2719 6.0415 5.1406 |  |  |  | |  |
| GAMMAs 1.2813 1.2947 1.3651 |  |  |  | |  |
| SON VEL,M/SEC 1779.4 1671.8 1062.4 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 3.466 |  |  |  | |  |
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| PERFORMANCE PARAMETERS |  |  |  | |  |
|  |  |  |  | |  |
| Ae/At 1.0000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2363.1 2363.1 |  |  |  | |  |
| CF 0.7075 1.5585 |  |  |  | |  |
| Ivac, M/SEC 2963.0 3907.3 |  |  |  | |  |
| Isp, M/SEC 1671.8 3682.7 |  |  |  | |  |
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| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H 0.00002 0.00000 0.00000 |  |  |  | |  |
| \*H2 0.24932 0.24933 0.24934 |  |  |  | |  |
| H2O 0.75066 0.75066 0.75066 |  |  |  | |  |
| \*OH 0.00001 0.00000 0.00000 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
|  |  |  |  | |  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
|  |  |  |  | |  |
| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 3.00000 %FUEL= 25.000000 R,EQ.RATIO= 2.645561 PHI,EQ.RATIO= 2.645561 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.7947 73.602 |  |  |  | |  |
| P, BAR 8.0000 4.4577 0.10869 |  |  |  | |  |
| T, K 2430.88 2185.45 971.32 |  |  |  | |  |
| RHO, KG/CU M 3.1824-1 1.9760-1 1.0852-2 |  |  |  | |  |
| H, KJ/KG -1421.83 -2815.97 -8643.89 |  |  |  | |  |
| U, KJ/KG -3935.65 -5071.83 -9645.44 |  |  |  | |  |
| G, KJ/KG -66034.5 -60905.0 -34461.5 |  |  |  | |  |
| S, KJ/(KG)(K) 26.5799 26.5799 26.5799 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 8.040 8.055 8.064 |  |  |  | |  |
| MW, MOL WT 8.040 8.055 8.064 |  |  |  | |  |
| (dLV/dLP)t -1.00145 -1.00053 -1.00000 |  |  |  | |  |
| (dLV/dLT)p 1.0337 1.0136 1.0000 |  |  |  | |  |
| Cp, KJ/(KG)(K) 6.1171 5.5380 4.2399 |  |  |  | |  |
| GAMMAs 1.2183 1.2360 1.3213 |  |  |  | |  |
| SON VEL,M/SEC 1750.0 1669.8 1150.4 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 3.304 |  |  |  | |  |
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| PERFORMANCE PARAMETERS |  |  |  | |  |
|  |  |  |  | |  |
| Ae/At 1.0000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2424.5 2424.5 |  |  |  | |  |
| CF 0.6887 1.5675 |  |  |  | |  |
| Ivac, M/SEC 3020.8 4064.1 |  |  |  | |  |
| Isp, M/SEC 1669.8 3800.5 |  |  |  | |  |
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| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H 0.00064 0.00024 0.00000 |  |  |  | |  |
| \*H2 0.15494 0.15528 0.15550 |  |  |  | |  |
| H2O 0.84301 0.84409 0.84450 |  |  |  | |  |
| \*O 0.00001 0.00000 0.00000 |  |  |  | |  |
| \*OH 0.00139 0.00038 0.00000 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
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| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 4.00000 %FUEL= 20.000000 R,EQ.RATIO= 1.984171 PHI,EQ.RATIO= 1.984171 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.7592 64.777 |  |  |  | |  |
| P, BAR 8.0000 4.5475 0.12350 |  |  |  | |  |
| T, K 2853.80 2644.42 1366.32 |  |  |  | |  |
| RHO, KG/CU M 3.3428-1 2.0646-1 1.0957-2 |  |  |  | |  |
| H, KJ/KG -1218.59 -2516.34 -8417.49 |  |  |  | |  |
| U, KJ/KG -3611.80 -4718.97 -9544.57 |  |  |  | |  |
| G, KJ/KG -67618.3 -64044.6 -40207.9 |  |  |  | |  |
| S, KJ/(KG)(K) 23.2672 23.2672 23.2672 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 9.915 9.982 10.079 |  |  |  | |  |
| MW, MOL WT 9.915 9.982 10.079 |  |  |  | |  |
| (dLV/dLP)t -1.00833 -1.00487 -1.00000 |  |  |  | |  |
| (dLV/dLT)p 1.1711 1.1070 1.0001 |  |  |  | |  |
| Cp, KJ/(KG)(K) 7.6031 6.5322 3.8550 |  |  |  | |  |
| GAMMAs 1.1668 1.1784 1.2723 |  |  |  | |  |
| SON VEL,M/SEC 1671.0 1611.1 1197.5 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 3.169 |  |  |  | |  |
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| PERFORMANCE PARAMETERS |  |  |  | |  |
|  |  |  |  | |  |
| Ae/At 1.0000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2405.1 2405.1 |  |  |  | |  |
| CF 0.6698 1.5776 |  |  |  | |  |
| Ivac, M/SEC 2978.2 4091.5 |  |  |  | |  |
| Isp, M/SEC 1611.1 3794.4 |  |  |  | |  |
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| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H 0.00246 0.00152 0.00000 |  |  |  | |  |
| \*H2 0.09763 0.09812 0.09920 |  |  |  | |  |
| H2O 0.88575 0.89323 0.90080 |  |  |  | |  |
| \*O 0.00036 0.00012 0.00000 |  |  |  | |  |
| \*OH 0.01350 0.00691 0.00000 |  |  |  | |  |
| \*O2 0.00030 0.00010 0.00000 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
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| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 5.00000 %FUEL= 16.666667 R,EQ.RATIO= 1.587337 PHI,EQ.RATIO= 1.587337 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.7369 57.120 |  |  |  | |  |
| P, BAR 8.0000 4.6060 0.14006 |  |  |  | |  |
| T, K 3102.15 2929.41 1772.77 |  |  |  | |  |
| RHO, KG/CU M 3.6018-1 2.2201-1 1.1490-2 |  |  |  | |  |
| H, KJ/KG -1083.09 -2268.59 -7998.80 |  |  |  | |  |
| U, KJ/KG -3304.20 -4343.26 -9217.76 |  |  |  | |  |
| G, KJ/KG -66005.7 -63575.9 -45099.8 |  |  |  | |  |
| S, KJ/(KG)(K) 20.9282 20.9282 20.9282 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 11.613 11.740 12.092 |  |  |  | |  |
| MW, MOL WT 11.613 11.740 12.092 |  |  |  | |  |
| (dLV/dLP)t -1.02148 -1.01559 -1.00013 |  |  |  | |  |
| (dLV/dLT)p 1.4182 1.3200 1.0041 |  |  |  | |  |
| Cp, KJ/(KG)(K) 10.0061 8.7785 3.7134 |  |  |  | |  |
| GAMMAs 1.1395 1.1428 1.2293 |  |  |  | |  |
| SON VEL,M/SEC 1590.9 1539.8 1224.1 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 3.038 |  |  |  | |  |
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| PERFORMANCE PARAMETERS |  |  |  | |  |
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| Ae/At 1.0000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2340.2 2340.2 |  |  |  | |  |
| CF 0.6580 1.5892 |  |  |  | |  |
| Ivac, M/SEC 2887.2 4046.8 |  |  |  | |  |
| Isp, M/SEC 1539.8 3719.1 |  |  |  | |  |
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| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H 0.00390 0.00302 0.00004 |  |  |  | |  |
| HO2 0.00001 0.00001 0.00000 |  |  |  | |  |
| \*H2 0.06103 0.06081 0.06164 |  |  |  | |  |
| H2O 0.88730 0.90379 0.93825 |  |  |  | |  |
| \*O 0.00278 0.00152 0.00000 |  |  |  | |  |
| \*OH 0.04143 0.02884 0.00008 |  |  |  | |  |
| \*O2 0.00355 0.00201 0.00000 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
|  |  |  |  | |  |
| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 6.00000 %FUEL= 14.285714 R,EQ.RATIO= 1.322780 PHI,EQ.RATIO= 1.322780 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.7258 51.077 |  |  |  | |  |
| P, BAR 8.0000 4.6354 0.15663 |  |  |  | |  |
| T, K 3233.14 3082.75 2152.83 |  |  |  | |  |
| RHO, KG/CU M 3.9072-1 2.4063-1 1.2299-2 |  |  |  | |  |
| H, KJ/KG -986.31 -2070.23 -7469.06 |  |  |  | |  |
| U, KJ/KG -3033.81 -3996.61 -8742.57 |  |  |  | |  |
| G, KJ/KG -63029.0 -61227.0 -48781.1 |  |  |  | |  |
| S, KJ/(KG)(K) 19.1896 19.1896 19.1896 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 13.129 13.306 14.055 |  |  |  | |  |
| MW, MOL WT 13.129 13.306 14.055 |  |  |  | |  |
| (dLV/dLP)t -1.03876 -1.03197 -1.00201 |  |  |  | |  |
| (dLV/dLT)p 1.7341 1.6350 1.0553 |  |  |  | |  |
| Cp, KJ/(KG)(K) 12.6107 11.6533 4.3422 |  |  |  | |  |
| GAMMAs 1.1265 1.1253 1.1761 |  |  |  | |  |
| SON VEL,M/SEC 1518.7 1472.4 1223.8 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 2.942 |  |  |  | |  |
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| PERFORMANCE PARAMETERS |  |  |  | |  |
|  |  |  |  | |  |
| Ae/At 1.0000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2258.0 2258.0 |  |  |  | |  |
| CF 0.6521 1.5947 |  |  |  | |  |
| Ivac, M/SEC 2780.7 3954.4 |  |  |  | |  |
| Isp, M/SEC 1472.4 3600.8 |  |  |  | |  |
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|  |  |  |  | |  |
| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H 0.00417 0.00353 0.00038 |  |  |  | |  |
| HO2 0.00005 0.00003 0.00000 |  |  |  | |  |
| \*H2 0.03825 0.03723 0.03467 |  |  |  | |  |
| H2O 0.85808 0.88043 0.96190 |  |  |  | |  |
| H2O2 0.00001 0.00001 0.00000 |  |  |  | |  |
| \*O 0.00843 0.00601 0.00004 |  |  |  | |  |
| \*OH 0.07452 0.06025 0.00291 |  |  |  | |  |
| \*O2 0.01650 0.01251 0.00010 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
|  |  |  |  | |  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
|  |  |  |  | |  |
| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 7.00000 %FUEL= 12.500000 R,EQ.RATIO= 1.133812 PHI,EQ.RATIO= 1.133812 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.7215 47.768 |  |  |  | |  |
| P, BAR 8.0000 4.6470 0.16748 |  |  |  | |  |
| T, K 3287.29 3146.55 2396.56 |  |  |  | |  |
| RHO, KG/CU M 4.2352-1 2.6074-1 1.3278-2 |  |  |  | |  |
| H, KJ/KG -913.72 -1910.46 -6919.74 |  |  |  | |  |
| U, KJ/KG -2802.65 -3692.66 -8181.09 |  |  |  | |  |
| G, KJ/KG -59578.2 -58063.2 -49688.5 |  |  |  | |  |
| S, KJ/(KG)(K) 17.8459 17.8459 17.8459 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 14.470 14.680 15.798 |  |  |  | |  |
| MW, MOL WT 14.470 14.680 15.798 |  |  |  | |  |
| (dLV/dLP)t -1.05216 -1.04634 -1.01177 |  |  |  | |  |
| (dLV/dLT)p 1.9741 1.9051 1.3037 |  |  |  | |  |
| Cp, KJ/(KG)(K) 13.9645 13.4953 7.3859 |  |  |  | |  |
| GAMMAs 1.1213 1.1185 1.1228 |  |  |  | |  |
| SON VEL,M/SEC 1455.4 1411.9 1190.0 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 2.912 |  |  |  | |  |
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| PERFORMANCE PARAMETERS |  |  |  | |  |
|  |  |  |  | |  |
| Ae/At 1.00000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2173.1 2173.1 |  |  |  | |  |
| CF 0.6497 1.5949 |  |  |  | |  |
| Ivac, M/SEC 2674.2 3829.8 |  |  |  | |  |
| Isp, M/SEC 1411.9 3465.8 |  |  |  | |  |
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|  |  |  |  | |  |
| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H 0.00366 0.00318 0.00087 |  |  |  | |  |
| HO2 0.00012 0.00008 0.00000 |  |  |  | |  |
| \*H2 0.02455 0.02319 0.01605 |  |  |  | |  |
| H2O 0.81205 0.83567 0.95550 |  |  |  | |  |
| H2O2 0.00002 0.00001 0.00000 |  |  |  | |  |
| \*O 0.01540 0.01238 0.00121 |  |  |  | |  |
| \*OH 0.09987 0.08636 0.01972 |  |  |  | |  |
| \*O2 0.04433 0.03914 0.00666 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |  |  |  | |  |
|  |  |  |  | |  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |  |  |  | |  |
|  |  |  |  | |  |
| Pin = 116.0 PSIA |  |  |  | |  |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |  |  |  | |  |
|  |  |  |  | |  |
| REACTANT WT FRACTION ENERGY TEMP |  |  |  | |  |
| (SEE NOTE) KJ/KG-MOL K |  |  |  | |  |
| FUEL H2(L) 1.0000000 -9012.000 20.270 |  |  |  | |  |
| OXIDANT O2(L) 1.0000000 -12979.000 90.170 |  |  |  | |  |
|  |  |  |  | |  |
| O/F= 8.00000 %FUEL= 11.111111 R,EQ.RATIO= 0.992085 PHI,EQ.RATIO= 0.992085 |  |  |  | |  |
|  |  |  |  | |  |
| CHAMBER THROAT EXIT |  |  |  | |  |
| Pinf/P 1.0000 1.7205 47.098 |  |  |  | |  |
| P, BAR 8.0000 4.6497 0.16986 |  |  |  | |  |
| T, K 3295.74 3157.26 2449.14 |  |  |  | |  |
| RHO, KG/CU M 4.5695-1 2.8129-1 1.4313-2 |  |  |  | |  |
| H, KJ/KG -857.26 -1780.40 -6428.64 |  |  |  | |  |
| U, KJ/KG -2608.01 -3433.39 -7615.41 |  |  |  | |  |
| G, KJ/KG -56146.7 -54746.6 -47515.4 |  |  |  | |  |
| S, KJ/(KG)(K) 16.7760 16.7760 16.7760 |  |  |  | |  |
|  |  |  |  | |  |
| M, (1/n) 15.652 15.881 17.159 |  |  |  | |  |
| MW, MOL WT 15.652 15.881 17.159 |  |  |  | |  |
| (dLV/dLP)t -1.05543 -1.05009 -1.02021 |  |  |  | |  |
| (dLV/dLT)p 2.0335 1.9759 1.5105 |  |  |  | |  |
| Cp, KJ/(KG)(K) 13.5129 13.2053 9.3160 |  |  |  | |  |
| GAMMAs 1.1200 1.1169 1.1092 |  |  |  | |  |
| SON VEL,M/SEC 1400.3 1358.8 1147.3 |  |  |  | |  |
| MACH NUMBER 0.000 1.000 2.909 |  |  |  | |  |
|  |  |  |  | |  |
| PERFORMANCE PARAMETERS |  |  |  | |  |
|  |  |  |  | |  |
| Ae/At 1.0000 8.0000 |  |  |  | |  |
| CSTAR, M/SEC 2093.1 2093.1 |  |  |  | |  |
| CF 0.6492 1.5948 |  |  |  | |  |
| Ivac, M/SEC 2575.3 3693.6 |  |  |  | |  |
| Isp, M/SEC 1358.8 3338.1 |  |  |  | |  |
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|  |  |  |  | |  |
| MASS FRACTIONS |  |  |  | |  |
|  |  |  |  | |  |
| \*H 0.00293 0.00253 0.00070 |  |  |  | |  |
| HO2 0.00019 0.00013 0.00001 |  |  |  | |  |
| \*H2 0.01632 0.01499 0.00709 |  |  |  | |  |
| H2O 0.76082 0.78314 0.90406 |  |  |  | |  |
| H2O2 0.00002 0.00001 0.00000 |  |  |  | |  |
| \*O 0.02105 0.01771 0.00406 |  |  |  | |  |
| \*OH 0.11333 0.10039 0.03641 |  |  |  | |  |
| \*O2 0.08534 0.08109 0.04768 |  |  |  | |  |
|  |  |  |  | |  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |  |  |  | |  |
|  |  |  |  | |  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |  |  |  | |  |
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| NASA-GLENN CHEMICAL EQUILIBRIUM PROGRAM CEA2, FEBRUARY 5, 2004 | | | |  | |
| BY BONNIE MCBRIDE AND SANFORD GORDON | | | |  | |
| REFS: NASA RP-1311, PART I, 1994 AND NASA RP-1311, PART II, 1996 | | | |  | |
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| ### CEA analysis performed on Sat 23-Apr-2022 16:51:06 | | | |  | |
|  | | | |  | |
| # Problem Type: "Rocket" (Infinite Area Combustor) | | | |  | |
|  | | | |  | |
| prob case=\_\_\_\_\_\_\_\_\_\_\_\_\_\_\_5064 ro equilibrium | | | |  | |
|  | | | |  | |
| # Pressure (1 value): | | | |  | |
| p,atm= 8 | | | |  | |
| # Supersonic Area Ratio (1 value): | | | |  | |
| supar= 8 | | | |  | |
|  | | | |  | |
| # Oxidizer/Fuel Wt. ratio (8 values): | | | |  | |
| o/f= 1, 2, 3, 4, 5, 6, 7, 8 | | | |  | |
|  | | | |  | |
| # You selected the following fuels and oxidizers: | | | |  | |
| reac | | | |  | |
| fuel paraffin wt%=100.0000 | | | |  | |
| oxid N2O wt%=100.0000 | | | |  | |
|  | | | |  | |
| # You selected these options for output: | | | |  | |
| # short version of output | | | |  | |
| output short | | | |  | |
| # Proportions of any products will be expressed as Mass Fractions. | | | |  | |
| output massf | | | |  | |
| # Heat will be expressed as siunits | | | |  | |
| output siunits | | | |  | |
|  | | | |  | |
| # Input prepared by this script:/var/www/sites/cearun.grc.nasa.gov/cgi-bin/CEARU | | | |  | |
| N/prepareInputFile.cgi | | | |  | |
|  | | | |  | |
| ### IMPORTANT: The following line is the end of your CEA input file! | | | |  | |
| end | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
|  | | | |  | |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
|  | | | |  | |
| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 1.00000 %FUEL= 50.000000 R,EQ.RATIO= 9.138527 PHI,EQ.RATIO= 9.138527 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.7293 49.334 | | | |  | |
| P, BAR 8.1060 4.6874 0.16431 | | | |  | |
| T, K 1027.92 976.19 721.80 | | | |  | |
| RHO, KG/CU M 2.1582 0 1.3309 0 6.8352-2 | | | |  | |
| H, KJ/KG -928.66 -1127.88 -2108.83 | | | |  | |
| U, KJ/KG -1304.24 -1480.08 -2349.21 | | | |  | |
| G, KJ/KG -10773.7 -10477.4 -9022.00 | | | |  | |
| S, KJ/(KG)(K) 9.5776 9.5776 9.5776 | | | |  | |
|  | | | |  | |
| M, (1/n) 22.756 23.045 24.966 | | | |  | |
| MW, MOL WT 14.290 14.263 14.449 | | | |  | |
| (dLV/dLP)t -1.08636 -1.08399 -1.06375 | | | |  | |
| (dLV/dLT)p 2.1411 2.1535 1.9897 | | | |  | |
| Cp, KJ/(KG)(K) 8.1621 8.3636 7.3598 | | | |  | |
| GAMMAs 1.1349 1.1313 1.1304 | | | |  | |
| SON VEL,M/SEC 652.9 631.2 521.3 | | | |  | |
| MACH NUMBER 0.000 1.000 2.947 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
|  | | | |  | |
| Ae/At 1.0000 8.0000 | | | |  | |
| CSTAR, M/SEC 964.9 964.9 | | | |  | |
| CF 0.6542 1.5922 | | | |  | |
| Ivac, M/SEC 1189.2 1692.8 | | | |  | |
| Isp, M/SEC 631.2 1536.3 | | | |  | |
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| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| CH4 0.07007 0.06906 0.07642 | | | |  | |
| \*CO 0.13177 0.10881 0.01517 | | | |  | |
| \*CO2 0.05839 0.06717 0.08698 | | | |  | |
| C2H6 0.00001 0.00000 0.00000 | | | |  | |
| \*H2 0.03663 0.03605 0.02932 | | | |  | |
| H2O 0.07210 0.07968 0.12369 | | | |  | |
| NH3 0.00040 0.00032 0.00009 | | | |  | |
| \*N2 0.31791 0.31798 0.31816 | | | |  | |
| C(gr) 0.31271 0.32092 0.35016 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
|  | | | |  | |
| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 2.00000 %FUEL= 33.333333 R,EQ.RATIO= 4.569263 PHI,EQ.RATIO= 4.569263 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.7433 49.215 | | | |  | |
| P, BAR 8.1060 4.6499 0.16470 | | | |  | |
| T, K 1161.32 1091.83 816.23 | | | |  | |
| RHO, KG/CU M 1.7639 0 1.0938 0 5.6961-2 | | | |  | |
| H, KJ/KG -619.11 -864.68 -2033.88 | | | |  | |
| U, KJ/KG -1078.66 -1289.81 -2323.04 | | | |  | |
| G, KJ/KG -12433.9 -11972.5 -10337.8 | | | |  | |
| S, KJ/(KG)(K) 10.1736 10.1736 10.1736 | | | |  | |
|  | | | |  | |
| M, (1/n) 21.012 21.353 23.470 | | | |  | |
| MW, MOL WT 17.263 17.266 17.174 | | | |  | |
| (dLV/dLP)t -1.05035 -1.05708 -1.05496 | | | |  | |
| (dLV/dLT)p 1.6690 1.8243 2.0250 | | | |  | |
| Cp, KJ/(KG)(K) 5.6069 6.7670 9.0574 | | | |  | |
| GAMMAs 1.1713 1.1553 1.1179 | | | |  | |
| SON VEL,M/SEC 733.7 700.8 568.5 | | | |  | |
| MACH NUMBER 0.000 1.000 2.959 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
|  | | | |  | |
| Ae/At 1.00000 8.0000 | | | |  | |
| CSTAR, M/SEC 1057.5 1057.5 | | | |  | |
| CF 0.6627 1.5907 | | | |  | |
| Ivac, M/SEC 1307.4 1854.0 | | | |  | |
| Isp, M/SEC 700.8 1682.1 | | | |  | |
|  | | | |  | |
|  | | | |  | |
| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| CH4 0.01707 0.01719 0.01474 | | | |  | |
| \*CO 0.33799 0.30616 0.11305 | | | |  | |
| \*CO2 0.03625 0.05296 0.16348 | | | |  | |
| HCN 0.00003 0.00001 0.00000 | | | |  | |
| \*H2 0.03438 0.03359 0.03045 | | | |  | |
| H2O 0.02581 0.03261 0.06633 | | | |  | |
| NH3 0.00018 0.00015 0.00004 | | | |  | |
| \*N2 0.42416 0.42419 0.42429 | | | |  | |
| C(gr) 0.12413 0.13314 0.18761 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
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| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 3.00000 %FUEL= 25.000000 R,EQ.RATIO= 3.046176 PHI,EQ.RATIO= 3.046176 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.7691 50.477 | | | |  | |
| P, BAR 8.1060 4.5820 0.16059 | | | |  | |
| T, K 1287.79 1177.76 855.70 | | | |  | |
| RHO, KG/CU M 1.5581 0 9.7621-1 5.2103-2 | | | |  | |
| H, KJ/KG -464.33 -745.98 -2009.21 | | | |  | |
| U, KJ/KG -984.59 -1215.34 -2317.42 | | | |  | |
| G, KJ/KG -13862.6 -12999.5 -10911.9 | | | |  | |
| S, KJ/(KG)(K) 10.4041 10.4041 10.4041 | | | |  | |
|  | | | |  | |
| M, (1/n) 20.581 20.863 23.084 | | | |  | |
| MW, MOL WT 19.951 19.995 19.981 | | | |  | |
| (dLV/dLP)t -1.01793 -1.02847 -1.05361 | | | |  | |
| (dLV/dLT)p 1.2177 1.3978 2.0893 | | | |  | |
| Cp, KJ/(KG)(K) 2.8174 3.9883 9.9546 | | | |  | |
| GAMMAs 1.2418 1.2001 1.1165 | | | |  | |
| SON VEL,M/SEC 803.8 750.5 586.6 | | | |  | |
| MACH NUMBER 0.000 1.000 2.996 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
|  | | | |  | |
| Ae/At 1.0000 8.0000 | | | |  | |
| CSTAR, M/SEC 1106.3 1106.3 | | | |  | |
| CF 0.6784 1.5888 | | | |  | |
| Ivac, M/SEC 1375.9 1933.1 | | | |  | |
| Isp, M/SEC 750.5 1757.8 | | | |  | |
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| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| CH4 0.00480 0.00572 0.00546 | | | |  | |
| \*CO 0.45154 0.42586 0.20771 | | | |  | |
| \*CO2 0.01144 0.02530 0.16432 | | | |  | |
| HCN 0.00009 0.00003 0.00000 | | | |  | |
| \*H2 0.02916 0.02835 0.02547 | | | |  | |
| H2O 0.00721 0.01237 0.03887 | | | |  | |
| NH3 0.00008 0.00008 0.00002 | | | |  | |
| \*N2 0.47724 0.47728 0.47735 | | | |  | |
| C(gr) 0.01843 0.02500 0.08081 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
|  | | | |  | |
| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 4.00000 %FUEL= 20.000000 R,EQ.RATIO= 2.284632 PHI,EQ.RATIO= 2.284632 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.8297 59.640 | | | |  | |
| P, BAR 8.1060 4.4302 0.13592 | | | |  | |
| T, K 1736.30 1514.26 858.15 | | | |  | |
| RHO, KG/CU M 1.2442 0 7.7973-1 4.4390-2 | | | |  | |
| H, KJ/KG -371.46 -739.38 -2145.17 | | | |  | |
| U, KJ/KG -1022.96 -1307.55 -2451.36 | | | |  | |
| G, KJ/KG -18438.0 -16495.5 -11074.4 | | | |  | |
| S, KJ/(KG)(K) 10.4052 10.4052 10.4052 | | | |  | |
|  | | | |  | |
| M, (1/n) 22.159 22.159 23.303 | | | |  | |
| MW, MOL WT 22.159 22.159 22.346 | | | |  | |
| (dLV/dLP)t -1.00004 -1.00004 -1.05215 | | | |  | |
| (dLV/dLT)p 1.0004 1.0003 2.1054 | | | |  | |
| Cp, KJ/(KG)(K) 1.6684 1.6475 10.1508 | | | |  | |
| GAMMAs 1.2904 1.2951 1.1156 | | | |  | |
| SON VEL,M/SEC 916.9 857.8 584.5 | | | |  | |
| MACH NUMBER 0.000 1.000 3.223 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
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| Ae/At 1.0000 8.0000 | | | |  | |
| CSTAR, M/SEC 1211.9 1211.9 | | | |  | |
| CF 0.7078 1.5541 | | | |  | |
| Ivac, M/SEC 1520.2 2046.0 | | | |  | |
| Isp, M/SEC 857.8 1883.5 | | | |  | |
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|  | | | |  | |
| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| CH4 0.00000 0.00000 0.00302 | | | |  | |
| \*CO 0.37995 0.37433 0.24006 | | | |  | |
| \*CO2 0.04441 0.05324 0.17502 | | | |  | |
| HCN 0.00001 0.00000 0.00000 | | | |  | |
| \*H2 0.01972 0.02013 0.02086 | | | |  | |
| H2O 0.04672 0.04311 0.02977 | | | |  | |
| NH3 0.00001 0.00001 0.00001 | | | |  | |
| \*N2 0.50917 0.50917 0.50918 | | | |  | |
| C(gr) 0.00000 0.00000 0.02208 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
|  | | | |  | |
| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 5.00000 %FUEL= 16.666667 R,EQ.RATIO= 1.827705 PHI,EQ.RATIO= 1.827705 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.8130 70.964 | | | |  | |
| P, BAR 8.1060 4.4710 0.11423 | | | |  | |
| T, K 2174.59 1920.86 879.75 | | | |  | |
| RHO, KG/CU M 1.0827 0 6.7623-1 3.7767-2 | | | |  | |
| H, KJ/KG -309.55 -728.54 -2408.37 | | | |  | |
| U, KJ/KG -1058.25 -1389.70 -2710.82 | | | |  | |
| G, KJ/KG -22527.2 -20353.8 -11396.7 | | | |  | |
| S, KJ/(KG)(K) 10.2169 10.2169 10.2169 | | | |  | |
|  | | | |  | |
| M, (1/n) 24.150 24.156 24.185 | | | |  | |
| MW, MOL WT 24.150 24.156 24.185 | | | |  | |
| (dLV/dLP)t -1.00018 -1.00005 -1.00217 | | | |  | |
| (dLV/dLT)p 1.0046 1.0013 1.0371 | | | |  | |
| Cp, KJ/(KG)(K) 1.6805 1.6350 1.8520 | | | |  | |
| GAMMAs 1.2604 1.2674 1.2461 | | | |  | |
| SON VEL,M/SEC 971.4 915.4 613.9 | | | |  | |
| MACH NUMBER 0.000 1.000 3.337 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
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| Ae/At 1.0000 8.0000 | | | |  | |
| CSTAR, M/SEC 1309.5 1309.5 | | | |  | |
| CF 0.6991 1.5646 | | | |  | |
| Ivac, M/SEC 1637.7 2196.4 | | | |  | |
| Isp, M/SEC 915.4 2048.8 | | | |  | |
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| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| CH4 0.00000 0.00000 0.00037 | | | |  | |
| \*CO 0.28876 0.28344 0.20649 | | | |  | |
| \*CO2 0.08080 0.08916 0.20906 | | | |  | |
| \*H 0.00003 0.00001 0.00000 | | | |  | |
| \*H2 0.01079 0.01119 0.01655 | | | |  | |
| H2O 0.08916 0.08579 0.03713 | | | |  | |
| NH3 0.00000 0.00000 0.00001 | | | |  | |
| \*NO 0.00001 0.00000 0.00000 | | | |  | |
| \*N2 0.53040 0.53040 0.53040 | | | |  | |
| \*OH 0.00006 0.00001 0.00000 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
|  | | | |  | |
| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 6.00000 %FUEL= 14.285714 R,EQ.RATIO= 1.523088 PHI,EQ.RATIO= 1.523088 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.7993 68.944 | | | |  | |
| P, BAR 8.1060 4.5052 0.11757 | | | |  | |
| T, K 2494.93 2231.94 1056.94 | | | |  | |
| RHO, KG/CU M 1.0071 0 6.2651-1 3.4547-2 | | | |  | |
| H, KJ/KG -265.33 -712.63 -2563.80 | | | |  | |
| U, KJ/KG -1070.18 -1431.73 -2904.13 | | | |  | |
| G, KJ/KG -25235.5 -23050.7 -13142.1 | | | |  | |
| S, KJ/(KG)(K) 10.0084 10.0084 10.0084 | | | |  | |
|  | | | |  | |
| M, (1/n) 25.774 25.807 25.822 | | | |  | |
| MW, MOL WT 25.774 25.807 25.822 | | | |  | |
| (dLV/dLP)t -1.00095 -1.00030 -1.00000 | | | |  | |
| (dLV/dLT)p 1.0234 1.0079 1.0000 | | | |  | |
| Cp, KJ/(KG)(K) 1.8008 1.6659 1.5654 | | | |  | |
| GAMMAs 1.2295 1.2441 1.2590 | | | |  | |
| SON VEL,M/SEC 994.8 945.8 654.6 | | | |  | |
| MACH NUMBER 0.000 1.000 3.276 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
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| Ae/At 1.0000 8.0000 | | | |  | |
| CSTAR, M/SEC 1367.9 1367.9 | | | |  | |
| CF 0.6914 1.5674 | | | |  | |
| Ivac, M/SEC 1706.1 2302.8 | | | |  | |
| Isp, M/SEC 945.8 2144.0 | | | |  | |
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| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| \*CO 0.21148 0.20747 0.14973 | | | |  | |
| \*CO2 0.12587 0.13217 0.22290 | | | |  | |
| \*H 0.00009 0.00003 0.00000 | | | |  | |
| \*H2 0.00529 0.00559 0.00976 | | | |  | |
| H2O 0.11076 0.10896 0.07206 | | | |  | |
| \*NO 0.00022 0.00004 0.00000 | | | |  | |
| \*N2 0.54545 0.54554 0.54556 | | | |  | |
| \*O 0.00001 0.00000 0.00000 | | | |  | |
| \*OH 0.00079 0.00020 0.00000 | | | |  | |
| \*O2 0.00003 0.00000 0.00000 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
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| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 7.00000 %FUEL= 12.500000 R,EQ.RATIO= 1.305504 PHI,EQ.RATIO= 1.305504 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.7800 67.007 | | | |  | |
| P, BAR 8.1060 4.5540 0.12097 | | | |  | |
| T, K 2704.90 2468.89 1212.86 | | | |  | |
| RHO, KG/CU M 9.7497-1 6.0255-1 3.2664-2 | | | |  | |
| H, KJ/KG -232.17 -689.80 -2665.42 | | | |  | |
| U, KJ/KG -1063.58 -1445.58 -3035.77 | | | |  | |
| G, KJ/KG -26771.9 -24913.9 -14565.6 | | | |  | |
| S, KJ/(KG)(K) 9.8117 9.8117 9.8117 | | | |  | |
|  | | | |  | |
| M, (1/n) 27.050 27.161 27.229 | | | |  | |
| MW, MOL WT 27.050 27.161 27.229 | | | |  | |
| (dLV/dLP)t -1.00371 -1.00136 -1.00000 | | | |  | |
| (dLV/dLT)p 1.0906 1.0357 1.0000 | | | |  | |
| Cp, KJ/(KG)(K) 2.2808 1.8701 1.4968 | | | |  | |
| GAMMAs 1.1857 1.2110 1.2563 | | | |  | |
| SON VEL,M/SEC 992.9 956.7 682.1 | | | |  | |
| MACH NUMBER 0.000 1.000 3.234 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
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| Ae/At 1.0000 8.0000 | | | |  | |
| CSTAR, M/SEC 1406.2 1406.2 | | | |  | |
| CF 0.6804 1.5688 | | | |  | |
| Ivac, M/SEC 1746.7 2373.9 | | | |  | |
| Isp, M/SEC 956.7 2206.0 | | | |  | |
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| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| \*CO 0.14166 0.13696 0.10145 | | | |  | |
| \*CO2 0.17830 0.18569 0.24148 | | | |  | |
| \*H 0.00014 0.00007 0.00000 | | | |  | |
| \*H2 0.00239 0.00246 0.00494 | | | |  | |
| H2O 0.11500 0.11601 0.09520 | | | |  | |
| \*NO 0.00181 0.00055 0.00000 | | | |  | |
| \*N2 0.55608 0.55667 0.55692 | | | |  | |
| \*O 0.00022 0.00004 0.00000 | | | |  | |
| \*OH 0.00347 0.00136 0.00000 | | | |  | |
| \*O2 0.00094 0.00019 0.00000 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM | | | |  | |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR | | | |  | |
|  | | | |  | |
| Pin = 116.0 PSIA | | | |  | |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ | | | |  | |
|  | | | |  | |
| REACTANT WT FRACTION ENERGY TEMP | | | |  | |
| (SEE NOTE) KJ/KG-MOL K | | | |  | |
| FUEL paraffin 1.0000000 -1860600.000 298.150 | | | |  | |
| OXIDANT N2O 1.0000000 0.000 0.000 | | | |  | |
|  | | | |  | |
| O/F= 8.00000 %FUEL= 11.111111 R,EQ.RATIO= 1.142316 PHI,EQ.RATIO= 1.142316 | | | |  | |
|  | | | |  | |
| CHAMBER THROAT EXIT | | | |  | |
| Pinf/P 1.0000 1.7492 64.054 | | | |  | |
| P, BAR 8.1060 4.6341 0.12655 | | | |  | |
| T, K 2784.43 2603.77 1362.43 | | | |  | |
| RHO, KG/CU M 9.7814-1 6.0285-1 3.1764-2 | | | |  | |
| H, KJ/KG -206.37 -652.94 -2719.68 | | | |  | |
| U, KJ/KG -1035.09 -1421.65 -3118.08 | | | |  | |
| G, KJ/KG -27033.5 -25739.5 -15846.3 | | | |  | |
| S, KJ/(KG)(K) 9.6347 9.6347 9.6347 | | | |  | |
|  | | | |  | |
| M, (1/n) 27.936 28.163 28.434 | | | |  | |
| MW, MOL WT 27.936 28.163 28.434 | | | |  | |
| (dLV/dLP)t -1.00921 -1.00552 -1.00000 | | | |  | |
| (dLV/dLT)p 1.2210 1.1426 1.0000 | | | |  | |
| Cp, KJ/(KG)(K) 3.1697 2.6612 1.4514 | | | |  | |
| GAMMAs 1.1505 1.1619 1.2523 | | | |  | |
| SON VEL,M/SEC 976.4 945.1 706.3 | | | |  | |
| MACH NUMBER 0.000 1.000 3.174 | | | |  | |
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| PERFORMANCE PARAMETERS | | | |  | |
|  | | | |  | |
| Ae/At 1.0000 8.0000 | | | |  | |
| CSTAR, M/SEC 1422.8 1422.8 | | | |  | |
| CF 0.6642 1.5758 | | | |  | |
| Ivac, M/SEC 1758.5 2419.7 | | | |  | |
| Isp, M/SEC 945.1 2242.0 | | | |  | |
|  | | | |  | |
|  | | | |  | |
| MASS FRACTIONS | | | |  | |
|  | | | |  | |
| \*CO 0.08795 0.07811 0.05443 | | | |  | |
| \*CO2 0.21815 0.23360 0.27081 | | | |  | |
| \*H 0.00012 0.00008 0.00000 | | | |  | |
| HO2 0.00001 0.00000 0.00000 | | | |  | |
| \*H2 0.00112 0.00102 0.00188 | | | |  | |
| H2O 0.10919 0.11194 0.10711 | | | |  | |
| \*NO 0.00560 0.00298 0.00000 | | | |  | |
| \*N2 0.56315 0.56437 0.56576 | | | |  | |
| \*O 0.00081 0.00034 0.00000 | | | |  | |
| \*OH 0.00682 0.00410 0.00000 | | | |  | |
| \*O2 0.00708 0.00345 0.00000 | | | |  | |
|  | | | |  | |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K | | | |  | |
|  | | | |  | |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS | | | |  | |

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| --- |
| REFS: NASA RP-1311, PART I, 1994 AND NASA RP-1311, PART II, 1996 |
|  |
| \*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\*\* |
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|  |
| ### CEA analysis performed on Sat 23-Apr-2022 17:18:29 |
|  |
| # Problem Type: "Rocket" (Infinite Area Combustor) |
|  |
| prob case=\_\_\_\_\_\_\_\_\_\_\_\_\_\_\_6844 ro equilibrium |
|  |
| # Pressure (1 value): |
| p,bar= 8 |
| # Supersonic Area Ratio (1 value): |
| supar= 8 |
|  |
| # Oxidizer/Fuel Wt. ratio (8 values): |
| o/f= 1, 2, 3, 4, 5, 6, 7, 8 |
|  |
| # You selected the following fuels and oxidizers: |
| reac |
| fuel AL wt%=100.0000 |
| oxid NH4CLO4(I) wt%=100.0000 |
|  |
| # You selected these options for output: |
| # short version of output |
| output short |
| # Proportions of any products will be expressed as Mass Fractions. |
| output massf |
| # Heat will be expressed as siunits |
| output siunits |
|  |
| # Input prepared by this script:/var/www/sites/cearun.grc.nasa.gov/cgi-bin/CEARU |
| N/prepareInputFile.cgi |
|  |
| ### IMPORTANT: The following line is the end of your CEA input file! |
| end |
|  |
|  |
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|  |
|  |
| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
|  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 1.00000 %FUEL= 50.000000 R,EQ.RATIO= 1.895919 PHI,EQ.RATIO= 2.612654 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7054 44.250 |
| P, BAR 8.0000 4.6910 0.18079 |
| T, K 3890.11 3758.97 3082.76 |
| RHO, KG/CU M 1.0813 0 6.6358-1 3.3214-2 |
| H, KJ/KG 0.00000 -386.04 -2407.71 |
| U, KJ/KG -739.86 -1092.97 -2952.04 |
| G, KJ/KG -29674.5 -29060.2 -25923.6 |
| S, KJ/(KG)(K) 7.6282 7.6282 7.6282 |
|  |
| M, (1/n) 43.717 44.211 47.089 |
| MW, MOL WT 38.134 38.215 38.819 |
| (dLV/dLP)t -1.14527 -1.13505 -1.08841 |
| (dLV/dLT)p 3.5665 3.4496 2.8088 |
| Cp, KJ/(KG)(K) 10.4503 10.1982 8.4081 |
| GAMMAs 1.0944 1.0922 1.0837 |
| SON VEL,M/SEC 899.8 878.7 768.1 |
| MACH NUMBER 0.000 1.000 2.857 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1372.0 1372.0 |
| CF 0.6404 1.5994 |
| Ivac, M/SEC 1683.2 2442.5 |
| Isp, M/SEC 878.7 2194.4 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.04105 0.04167 0.04487 |
| ALCL 0.16504 0.16903 0.19747 |
| ALCL2 0.00195 0.00163 0.00052 |
| ALCL3 0.00005 0.00004 0.00001 |
| ALH 0.00239 0.00206 0.00087 |
| ALHCL 0.00028 0.00021 0.00003 |
| ALHCL2 0.00003 0.00002 0.00000 |
| ALH2 0.00001 0.00000 0.00000 |
| ALN 0.00005 0.00003 0.00000 |
| \*ALO 0.03907 0.03467 0.01319 |
| ALOCL 0.00658 0.00579 0.00213 |
| ALOCL2 0.00001 0.00000 0.00000 |
| ALOH 0.08876 0.08248 0.04582 |
| ALOHCL 0.00227 0.00170 0.00024 |
| ALOHCL2 0.00019 0.00014 0.00001 |
| ALO2 0.00053 0.00035 0.00002 |
| AL(OH)2 0.00051 0.00034 0.00002 |
| AL(OH)2CL 0.00005 0.00003 0.00000 |
| AL(OH)3 0.00001 0.00001 0.00000 |
| AL2 0.00012 0.00009 0.00001 |
| AL2O 0.13788 0.13406 0.10917 |
| AL2O2 0.02332 0.01952 0.00507 |
| AL2O3 0.00008 0.00005 0.00000 |
| \*CL 0.02153 0.02099 0.01489 |
| CLO 0.00002 0.00001 0.00000 |
| CL2 0.00001 0.00001 0.00000 |
| \*H 0.00445 0.00457 0.00491 |
| HALO 0.00018 0.00013 0.00002 |
| HALO2 0.00021 0.00016 0.00002 |
| HCL 0.03070 0.02989 0.02301 |
| \*H2 0.00799 0.00824 0.00994 |
| H2O 0.01137 0.01027 0.00435 |
| \*N 0.00004 0.00003 0.00001 |
| \*NH 0.00001 0.00001 0.00000 |
| \*NO 0.00142 0.00112 0.00018 |
| \*N2 0.05888 0.05904 0.05952 |
| \*O 0.00394 0.00335 0.00080 |
| \*OH 0.00709 0.00603 0.00155 |
| \*O2 0.00047 0.00036 0.00003 |
| AL2O3(L) 0.34149 0.36185 0.46130 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
|  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 2.00000 %FUEL= 33.333333 R,EQ.RATIO= 1.170182 PHI,EQ.RATIO= 1.306327 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7056 44.163 |
| P, BAR 8.0000 4.6905 0.18115 |
| T, K 3974.32 3844.45 3184.22 |
| RHO, KG/CU M 8.9873-1 5.5161-1 2.7615-2 |
| H, KJ/KG 0.00000 -464.50 -2895.75 |
| U, KJ/KG -890.14 -1314.83 -3551.70 |
| G, KJ/KG -34163.0 -33511.2 -30267.1 |
| S, KJ/(KG)(K) 8.5959 8.5959 8.5959 |
|  |
| M, (1/n) 37.123 37.591 40.361 |
| MW, MOL WT 32.296 32.406 33.165 |
| (dLV/dLP)t -1.27178 -1.25690 -1.18110 |
| (dLV/dLT)p 5.6901 5.5573 4.6611 |
| Cp, KJ/(KG)(K) 20.2263 19.9975 17.4141 |
| GAMMAs 1.0950 1.0925 1.0822 |
| SON VEL,M/SEC 987.3 963.9 842.5 |
| MACH NUMBER 0.000 1.000 2.856 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1504.7 1504.7 |
| CF 0.6406 1.5994 |
| Ivac, M/SEC 1846.1 2679.1 |
| Isp, M/SEC 963.8 2406.6 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.00780 0.00712 0.00299 |
| ALCL 0.06708 0.06359 0.03681 |
| ALCL2 0.00193 0.00155 0.00034 |
| ALCL3 0.00010 0.00008 0.00002 |
| ALH 0.00037 0.00028 0.00004 |
| ALHCL 0.00010 0.00007 0.00001 |
| ALHCL2 0.00003 0.00002 0.00000 |
| ALN 0.00001 0.00001 0.00000 |
| \*ALO 0.03336 0.02930 0.01033 |
| ALOCL 0.01212 0.01089 0.00477 |
| ALOCL2 0.00004 0.00003 0.00000 |
| ALOH 0.05428 0.04896 0.02110 |
| ALOHCL 0.00341 0.00258 0.00040 |
| ALOHCL2 0.00063 0.00047 0.00007 |
| ALO2 0.00222 0.00162 0.00019 |
| AL(OH)2 0.00115 0.00082 0.00009 |
| AL(OH)2CL 0.00026 0.00018 0.00002 |
| AL(OH)3 0.00008 0.00006 0.00000 |
| AL2O 0.01321 0.01109 0.00247 |
| AL2O2 0.01012 0.00806 0.00137 |
| AL2O3 0.00017 0.00011 0.00001 |
| \*CL 0.07602 0.07844 0.09200 |
| CLO 0.00029 0.00023 0.00005 |
| CL2 0.00008 0.00007 0.00002 |
| \*H 0.00521 0.00540 0.00624 |
| HALO 0.00012 0.00009 0.00001 |
| HALO2 0.00062 0.00049 0.00010 |
| HCL 0.07990 0.08093 0.08799 |
| HNO 0.00001 0.00000 0.00000 |
| HOCL 0.00003 0.00002 0.00000 |
| HO2 0.00004 0.00003 0.00000 |
| \*H2 0.00688 0.00703 0.00776 |
| H2O 0.04392 0.04324 0.04002 |
| \*N 0.00007 0.00005 0.00001 |
| \*NH 0.00001 0.00001 0.00000 |
| \*NO 0.00978 0.00865 0.00392 |
| \*N2 0.07483 0.07537 0.07764 |
| \*O 0.02902 0.02799 0.02080 |
| \*OH 0.03859 0.03648 0.02484 |
| \*O2 0.01562 0.01460 0.00939 |
| AL2O3(L) 0.41048 0.43398 0.54817 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
|  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 3.00000 %FUEL= 25.000000 R,EQ.RATIO= 0.928269 PHI,EQ.RATIO= 0.870885 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7104 45.327 |
| P, BAR 8.0000 4.6772 0.17649 |
| T, K 3853.81 3717.96 3018.81 |
| RHO, KG/CU M 8.5704-1 5.2639-1 2.6497-2 |
| H, KJ/KG 0.00000 -488.84 -3014.53 |
| U, KJ/KG -933.44 -1377.39 -3680.63 |
| G, KJ/KG -34741.3 -34005.5 -30228.5 |
| S, KJ/(KG)(K) 9.0148 9.0148 9.0148 |
|  |
| M, (1/n) 34.327 34.790 37.682 |
| MW, MOL WT 30.408 30.612 32.216 |
| (dLV/dLP)t -1.17518 -1.16320 -1.11306 |
| (dLV/dLT)p 3.9703 3.8352 3.2540 |
| Cp, KJ/(KG)(K) 14.2620 13.8195 11.8087 |
| GAMMAs 1.1020 1.1003 1.0927 |
| SON VEL,M/SEC 1014.2 988.8 853.1 |
| MACH NUMBER 0.000 1.000 2.878 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1537.0 1537.0 |
| CF 0.6433 1.5975 |
| Ivac, M/SEC 1887.4 2726.7 |
| Isp, M/SEC 988.8 2455.4 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.00165 0.00128 0.00014 |
| ALCL 0.02668 0.02259 0.00494 |
| ALCL2 0.00120 0.00088 0.00009 |
| ALCL3 0.00011 0.00008 0.00001 |
| ALH 0.00008 0.00005 0.00000 |
| ALHCL 0.00003 0.00002 0.00000 |
| ALHCL2 0.00002 0.00001 0.00000 |
| \*ALO 0.01383 0.01093 0.00156 |
| ALOCL 0.00934 0.00790 0.00201 |
| ALOCL2 0.00004 0.00003 0.00000 |
| ALOH 0.02619 0.02173 0.00453 |
| ALOHCL 0.00254 0.00181 0.00016 |
| ALOHCL2 0.00084 0.00061 0.00007 |
| ALO2 0.00158 0.00108 0.00007 |
| AL(OH)2 0.00103 0.00071 0.00006 |
| AL(OH)2CL 0.00042 0.00030 0.00003 |
| AL(OH)3 0.00016 0.00011 0.00001 |
| AL2O 0.00181 0.00124 0.00005 |
| AL2O2 0.00269 0.00184 0.00009 |
| AL2O3 0.00008 0.00005 0.00000 |
| \*CL 0.09508 0.09695 0.10056 |
| CLO 0.00058 0.00047 0.00012 |
| CL2 0.00016 0.00013 0.00004 |
| \*H 0.00408 0.00408 0.00348 |
| HALO 0.00005 0.00003 0.00000 |
| HALO2 0.00054 0.00041 0.00006 |
| HCL 0.11151 0.11359 0.12518 |
| HNO 0.00001 0.00001 0.00000 |
| HOCL 0.00007 0.00005 0.00001 |
| HO2 0.00009 0.00007 0.00001 |
| \*H2 0.00603 0.00606 0.00569 |
| H2O 0.07583 0.07751 0.09496 |
| \*N 0.00005 0.00004 0.00000 |
| \*NH 0.00001 0.00000 0.00000 |
| \*NO 0.01519 0.01365 0.00689 |
| NO2 0.00001 0.00001 0.00000 |
| \*N2 0.08226 0.08300 0.08619 |
| \*O 0.03816 0.03665 0.02582 |
| \*OH 0.05653 0.05413 0.03997 |
| \*O2 0.04064 0.03993 0.03808 |
| AL2O3(L) 0.38279 0.39998 0.45912 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
|  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 4.00000 %FUEL= 20.000000 R,EQ.RATIO= 0.807313 PHI,EQ.RATIO= 0.653164 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7144 45.962 |
| P, BAR 8.0000 4.6663 0.17406 |
| T, K 3730.45 3589.73 2874.31 |
| RHO, KG/CU M 8.4657-1 5.2033-1 2.6307-2 |
| H, KJ/KG 0.00000 -496.28 -3033.52 |
| U, KJ/KG -944.99 -1393.10 -3695.15 |
| G, KJ/KG -34478.8 -33674.5 -29599.5 |
| S, KJ/(KG)(K) 9.2425 9.2425 9.2425 |
|  |
| M, (1/n) 32.822 33.281 36.121 |
| MW, MOL WT 29.620 29.898 31.889 |
| (dLV/dLP)t -1.12065 -1.11175 -1.07201 |
| (dLV/dLT)p 3.0389 2.9400 2.4929 |
| Cp, KJ/(KG)(K) 10.7150 10.3705 8.9347 |
| GAMMAs 1.1082 1.1068 1.0966 |
| SON VEL,M/SEC 1023.4 996.3 851.8 |
| MACH NUMBER 0.000 1.000 2.892 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1543.2 1543.2 |
| CF 0.6456 1.5961 |
| Ivac, M/SEC 1896.4 2731.7 |
| Isp, M/SEC 996.3 2463.1 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.00042 0.00028 0.00001 |
| ALCL 0.01171 0.00898 0.00103 |
| ALCL2 0.00074 0.00050 0.00003 |
| ALCL3 0.00011 0.00008 0.00001 |
| ALH 0.00002 0.00001 0.00000 |
| ALHCL 0.00001 0.00001 0.00000 |
| ALHCL2 0.00001 0.00001 0.00000 |
| \*ALO 0.00576 0.00410 0.00029 |
| ALOCL 0.00655 0.00522 0.00090 |
| ALOCL2 0.00004 0.00002 0.00000 |
| ALOH 0.01288 0.00981 0.00111 |
| ALOHCL 0.00173 0.00116 0.00007 |
| ALOHCL2 0.00093 0.00066 0.00007 |
| ALO2 0.00092 0.00058 0.00002 |
| AL(OH)2 0.00077 0.00051 0.00003 |
| AL(OH)2CL 0.00052 0.00036 0.00004 |
| AL(OH)3 0.00023 0.00016 0.00002 |
| AL2O 0.00033 0.00019 0.00000 |
| AL2O2 0.00078 0.00046 0.00001 |
| AL2O3 0.00003 0.00002 0.00000 |
| \*CL 0.09994 0.10053 0.09635 |
| CLO 0.00079 0.00063 0.00016 |
| CL2 0.00022 0.00018 0.00006 |
| \*H 0.00301 0.00289 0.00183 |
| HALO 0.00002 0.00001 0.00000 |
| HALO2 0.00039 0.00028 0.00003 |
| HCL 0.13249 0.13497 0.14786 |
| HNO 0.00001 0.00001 0.00000 |
| HOCL 0.00010 0.00008 0.00002 |
| HO2 0.00013 0.00010 0.00002 |
| \*H2 0.00505 0.00494 0.00372 |
| H2O 0.10321 0.10713 0.13706 |
| H2O2 0.00001 0.00000 0.00000 |
| \*N 0.00003 0.00002 0.00000 |
| \*NO 0.01822 0.01639 0.00813 |
| NO2 0.00002 0.00001 0.00000 |
| \*N2 0.08682 0.08769 0.09158 |
| \*O 0.03835 0.03613 0.02180 |
| \*OH 0.06411 0.06104 0.04144 |
| \*O2 0.06674 0.06712 0.07172 |
| AL2O3(L) 0.33585 0.34672 0.37462 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
|  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 5.00000 %FUEL= 16.666667 R,EQ.RATIO= 0.734739 PHI,EQ.RATIO= 0.522531 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7169 46.373 |
| P, BAR 8.0000 4.6594 0.17251 |
| T, K 3625.13 3481.51 2754.32 |
| RHO, KG/CU M 8.4583-1 5.2015-1 2.6370-2 |
| H, KJ/KG 0.00000 -497.59 -3024.98 |
| U, KJ/KG -945.82 -1393.38 -3679.18 |
| G, KJ/KG -34013.4 -33163.4 -28867.8 |
| S, KJ/(KG)(K) 9.3827 9.3827 9.3827 |
|  |
| M, (1/n) 31.868 32.315 35.006 |
| MW, MOL WT 29.201 29.516 31.601 |
| (dLV/dLP)t -1.09102 -1.08352 -1.04653 |
| (dLV/dLT)p 2.5519 2.4698 2.0080 |
| Cp, KJ/(KG)(K) 8.8392 8.5588 6.9657 |
| GAMMAs 1.1126 1.1109 1.1001 |
| SON VEL,M/SEC 1025.8 997.6 848.3 |
| MACH NUMBER 0.000 1.000 2.899 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1541.7 1541.7 |
| CF 0.6470 1.5954 |
| Ivac, M/SEC 1895.5 2725.6 |
| Isp, M/SEC 997.6 2459.7 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.00013 0.00008 0.00000 |
| ALCL 0.00584 0.00415 0.00028 |
| ALCL2 0.00048 0.00031 0.00001 |
| ALCL3 0.00011 0.00008 0.00001 |
| ALH 0.00001 0.00000 0.00000 |
| ALHCL 0.00001 0.00000 0.00000 |
| ALHCL2 0.00001 0.00000 0.00000 |
| \*ALO 0.00265 0.00173 0.00006 |
| ALOCL 0.00468 0.00356 0.00044 |
| ALOCL2 0.00003 0.00002 0.00000 |
| ALOH 0.00685 0.00484 0.00031 |
| ALOHCL 0.00119 0.00076 0.00003 |
| ALOHCL2 0.00096 0.00067 0.00007 |
| ALO2 0.00054 0.00031 0.00001 |
| AL(OH)2 0.00056 0.00035 0.00001 |
| AL(OH)2CL 0.00056 0.00039 0.00003 |
| AL(OH)3 0.00027 0.00019 0.00002 |
| AL2O 0.00008 0.00004 0.00000 |
| AL2O2 0.00026 0.00014 0.00000 |
| AL2O3 0.00001 0.00001 0.00000 |
| \*CL 0.10009 0.09971 0.09056 |
| CLO 0.00093 0.00074 0.00018 |
| CL2 0.00027 0.00022 0.00008 |
| \*H 0.00224 0.00207 0.00099 |
| HALO 0.00001 0.00000 0.00000 |
| HALO2 0.00028 0.00019 0.00001 |
| HCL 0.14723 0.14992 0.16480 |
| HNO 0.00001 0.00001 0.00000 |
| HOCL 0.00012 0.00010 0.00002 |
| HO2 0.00015 0.00011 0.00002 |
| \*H2 0.00419 0.00399 0.00240 |
| H2O 0.12464 0.12993 0.16462 |
| H2O2 0.00001 0.00001 0.00000 |
| \*N 0.00002 0.00001 0.00000 |
| \*NO 0.01989 0.01782 0.00843 |
| NO2 0.00002 0.00002 0.00000 |
| \*N2 0.09003 0.09100 0.09541 |
| \*O 0.03567 0.03292 0.01671 |
| \*OH 0.06606 0.06217 0.03753 |
| \*O2 0.09075 0.09228 0.10312 |
| AL2O3(L) 0.29216 0.29911 0.31384 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
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| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 6.00000 %FUEL= 14.285714 R,EQ.RATIO= 0.686357 PHI,EQ.RATIO= 0.435442 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7187 46.761 |
| P, BAR 8.0000 4.6546 0.17108 |
| T, K 3537.92 3392.33 2650.38 |
| RHO, KG/CU M 8.4845-1 5.2196-1 2.6511-2 |
| H, KJ/KG 0.00000 -496.64 -3008.11 |
| U, KJ/KG -942.89 -1388.40 -3653.44 |
| G, KJ/KG -33526.1 -32643.0 -28123.7 |
| S, KJ/(KG)(K) 9.4762 9.4762 9.4762 |
|  |
| M, (1/n) 31.198 31.629 34.148 |
| MW, MOL WT 28.931 29.262 31.321 |
| (dLV/dLP)t -1.07279 -1.06595 -1.03094 |
| (dLV/dLT)p 2.2587 2.1820 1.6982 |
| Cp, KJ/(KG)(K) 7.7051 7.4430 5.5940 |
| GAMMAs 1.1157 1.1138 1.1045 |
| SON VEL,M/SEC 1025.7 996.6 844.2 |
| MACH NUMBER 0.000 1.000 2.905 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1537.9 1537.9 |
| CF 0.6481 1.5949 |
| Ivac, M/SEC 1891.4 2715.9 |
| Isp, M/SEC 996.6 2452.8 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.00005 0.00003 0.00000 |
| ALCL 0.00326 0.00218 0.00009 |
| ALCL2 0.00034 0.00021 0.00001 |
| ALCL3 0.00010 0.00007 0.00001 |
| \*ALO 0.00135 0.00082 0.00002 |
| ALOCL 0.00348 0.00255 0.00023 |
| ALOCL2 0.00003 0.00002 0.00000 |
| ALOH 0.00395 0.00262 0.00010 |
| ALOHCL 0.00085 0.00052 0.00001 |
| ALOHCL2 0.00097 0.00067 0.00006 |
| ALO2 0.00033 0.00018 0.00000 |
| AL(OH)2 0.00041 0.00025 0.00001 |
| AL(OH)2CL 0.00059 0.00040 0.00003 |
| AL(OH)3 0.00029 0.00020 0.00002 |
| AL2O 0.00002 0.00001 0.00000 |
| AL2O2 0.00011 0.00005 0.00000 |
| AL2O3 0.00001 0.00000 0.00000 |
| \*CL 0.09869 0.09763 0.08429 |
| CLO 0.00103 0.00082 0.00019 |
| CL2 0.00031 0.00026 0.00010 |
| \*H 0.00172 0.00154 0.00055 |
| HALO2 0.00020 0.00013 0.00001 |
| HCL 0.15824 0.16116 0.17883 |
| HNO 0.00001 0.00000 0.00000 |
| HOCL 0.00014 0.00011 0.00002 |
| HO2 0.00017 0.00013 0.00002 |
| \*H2 0.00351 0.00327 0.00156 |
| H2O 0.14100 0.14702 0.18274 |
| H2O2 0.00001 0.00001 0.00000 |
| \*N 0.00001 0.00001 0.00000 |
| \*NO 0.02077 0.01852 0.00824 |
| NOCL 0.00001 0.00000 0.00000 |
| NO2 0.00003 0.00002 0.00000 |
| \*N2 0.09246 0.09352 0.09834 |
| \*O 0.03242 0.02933 0.01232 |
| \*OH 0.06535 0.06072 0.03216 |
| \*O2 0.11173 0.11426 0.13056 |
| AL2O3(L) 0.25604 0.26075 0.26949 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
|  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 7.00000 %FUEL= 12.500000 R,EQ.RATIO= 0.651798 PHI,EQ.RATIO= 0.373236 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7201 47.178 |
| P, BAR 8.0000 4.6508 0.16957 |
| T, K 3465.46 3318.25 2558.12 |
| RHO, KG/CU M 8.5228-1 5.2446-1 2.6676-2 |
| H, KJ/KG 0.00000 -494.89 -2988.85 |
| U, KJ/KG -938.65 -1381.67 -3624.50 |
| G, KJ/KG -33068.9 -32159.0 -27399.5 |
| S, KJ/(KG)(K) 9.5424 9.5424 9.5424 |
|  |
| M, (1/n) 30.697 31.112 33.461 |
| MW, MOL WT 28.734 29.070 31.056 |
| (dLV/dLP)t -1.06056 -1.05411 -1.02128 |
| (dLV/dLT)p 2.0633 1.9881 1.4978 |
| Cp, KJ/(KG)(K) 6.9416 6.6776 4.6336 |
| GAMMAs 1.1180 1.1162 1.1099 |
| SON VEL,M/SEC 1024.4 994.9 840.0 |
| MACH NUMBER 0.000 1.000 2.911 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1533.2 1533.2 |
| CF 0.6489 1.5946 |
| Ivac, M/SEC 1886.2 2704.9 |
| Isp, M/SEC 994.9 2444.9 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.00002 0.00001 0.00000 |
| ALCL 0.00199 0.00126 0.00003 |
| ALCL2 0.00025 0.00015 0.00000 |
| ALCL3 0.00010 0.00007 0.00001 |
| \*ALO 0.00076 0.00043 0.00000 |
| ALOCL 0.00268 0.00190 0.00012 |
| ALOCL2 0.00002 0.00001 0.00000 |
| ALOH 0.00245 0.00154 0.00003 |
| ALOHCL 0.00064 0.00037 0.00001 |
| ALOHCL2 0.00097 0.00067 0.00006 |
| ALO2 0.00021 0.00011 0.00000 |
| AL(OH)2 0.00031 0.00018 0.00000 |
| AL(OH)2CL 0.00059 0.00040 0.00003 |
| AL(OH)3 0.00030 0.00020 0.00001 |
| AL2O 0.00001 0.00000 0.00000 |
| AL2O2 0.00005 0.00002 0.00000 |
| \*CL 0.09682 0.09523 0.07791 |
| CLO 0.00110 0.00087 0.00020 |
| CL2 0.00036 0.00030 0.00013 |
| \*H 0.00135 0.00118 0.00031 |
| HALO2 0.00015 0.00009 0.00000 |
| HCL 0.16688 0.17004 0.19100 |
| HNO 0.00001 0.00000 0.00000 |
| HOCL 0.00016 0.00013 0.00003 |
| HO2 0.00018 0.00013 0.00002 |
| \*H2 0.00299 0.00272 0.00103 |
| H2O 0.15356 0.15994 0.19492 |
| H2O2 0.00001 0.00001 0.00000 |
| \*N 0.00001 0.00001 0.00000 |
| \*NO 0.02121 0.01880 0.00779 |
| NOCL 0.00001 0.00000 0.00000 |
| NO2 0.00003 0.00002 0.00000 |
| \*N2 0.09439 0.09552 0.10068 |
| \*O 0.02933 0.02606 0.00893 |
| \*OH 0.06349 0.05826 0.02683 |
| \*O2 0.12973 0.13309 0.15395 |
| AL2O3(L) 0.22685 0.23025 0.23598 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |
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| THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM |
|  |
| COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR |
|  |
| Pin = 116.0 PSIA |
| CASE = \_\_\_\_\_\_\_\_\_\_\_\_\_\_\_ |
|  |
| REACTANT WT FRACTION ENERGY TEMP |
| (SEE NOTE) KJ/KG-MOL K |
| FUEL AL 1.0000000 0.000 0.000 |
| OXIDANT NH4CLO4(I) 1.0000000 0.000 0.000 |
|  |
| O/F= 8.00000 %FUEL= 11.111111 R,EQ.RATIO= 0.625879 PHI,EQ.RATIO= 0.326582 |
|  |
| CHAMBER THROAT EXIT |
| Pinf/P 1.0000 1.7214 47.630 |
| P, BAR 8.0000 4.6474 0.16796 |
| T, K 3404.56 3255.85 2475.25 |
| RHO, KG/CU M 8.5648-1 5.2715-1 2.6847-2 |
| H, KJ/KG 0.00000 -492.90 -2969.36 |
| U, KJ/KG -934.05 -1374.51 -3594.99 |
| G, KJ/KG -32654.4 -31721.0 -26710.4 |
| S, KJ/(KG)(K) 9.5914 9.5914 9.5914 |
|  |
| M, (1/n) 30.306 30.706 32.896 |
| MW, MOL WT 28.580 28.914 30.810 |
| (dLV/dLP)t -1.05186 -1.04571 -1.01516 |
| (dLV/dLT)p 1.9241 1.8493 1.3657 |
| Cp, KJ/(KG)(K) 6.3890 6.1162 3.9526 |
| GAMMAs 1.1200 1.1182 1.1162 |
| SON VEL,M/SEC 1022.8 992.9 835.7 |
| MACH NUMBER 0.000 1.000 2.916 |
|  |
| PERFORMANCE PARAMETERS |
|  |
| Ae/At 1.0000 8.0000 |
| CSTAR, M/SEC 1528.5 1528.5 |
| CF 0.6496 1.5944 |
| Ivac, M/SEC 1880.8 2693.7 |
| Isp, M/SEC 992.9 2436.9 |
|  |
|  |
| MASS FRACTIONS |
|  |
| \*AL 0.00001 0.00000 0.00000 |
| ALCL 0.00130 0.00079 0.00001 |
| ALCL2 0.00019 0.00011 0.00000 |
| ALCL3 0.00010 0.00007 0.00001 |
| \*ALO 0.00046 0.00024 0.00000 |
| ALOCL 0.00214 0.00147 0.00007 |
| ALOCL2 0.00002 0.00001 0.00000 |
| ALOH 0.00162 0.00096 0.00001 |
| ALOHCL 0.00049 0.00028 0.00000 |
| ALOHCL2 0.00097 0.00066 0.00005 |
| ALO2 0.00014 0.00007 0.00000 |
| AL(OH)2 0.00024 0.00013 0.00000 |
| AL(OH)2CL 0.00059 0.00039 0.00002 |
| AL(OH)3 0.00030 0.00020 0.00001 |
| AL2O2 0.00002 0.00001 0.00000 |
| \*CL 0.09485 0.09282 0.07166 |
| CLO 0.00115 0.00091 0.00020 |
| CL2 0.00040 0.00034 0.00016 |
| \*H 0.00110 0.00093 0.00018 |
| HALO2 0.00012 0.00007 0.00000 |
| HCL 0.17390 0.17731 0.20175 |
| HNO 0.00001 0.00000 0.00000 |
| HOCL 0.00018 0.00014 0.00003 |
| HO2 0.00018 0.00013 0.00002 |
| \*H2 0.00259 0.00231 0.00069 |
| H2O 0.16338 0.16991 0.20326 |
| H2O2 0.00001 0.00001 0.00000 |
| \*N 0.00001 0.00000 0.00000 |
| \*NO 0.02138 0.01885 0.00724 |
| NOCL 0.00001 0.00000 0.00000 |
| NO2 0.00003 0.00003 0.00000 |
| \*N2 0.09596 0.09716 0.10259 |
| \*O 0.02662 0.02324 0.00642 |
| \*OH 0.06121 0.05550 0.02209 |
| \*O2 0.14512 0.14914 0.17371 |
| AL2O3(L) 0.20318 0.20578 0.20982 |
|  |
| \* THERMODYNAMIC PROPERTIES FITTED TO 20000.K |
|  |
| NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS |