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J. Bradford
J. Olds
Georgia Institute of Technology
Atlanta, GA

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SCCREAM v.5 : A Web-Based Airbreathing Propulsion Analysis Tool

John E. Bradford†
Dr. John R. Olds*
Space Systems Design Laboratory
School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA 30332-0150

ABSTRACT

To properly assess the advantages and disadvantages of various RBCC design options at the conceptual vehicle level, an engine performance analysis tool is required. This tool must be capable of modeling engine performance effects that will subsequently be propagated throughout the conceptual design process via trajectory analysis, weight assessment, fuel balance calculations, thermal environment, life cycle cost, etc. For a given engine configuration, the tool will need to generate engine thrust and Isp as a function of altitude and Mach number for each operating mode of an RBCC engine.

A project to create a new computer program for the analysis of RBCC engines has already been initiated. Called SCCREAM, for Simulated Combined-Cycle Rocket Engine Analysis Module, it is intended for use in the conceptual phase of airbreathing launch vehicle design.

This paper will detail the capabilities of the latest version of SCCREAM and present the results of validation efforts. Combustor thermodynamic properties and overall engine performance for a sample engine will be compared with industry standard codes. Results from the new scram-rocket mode will be discussed. Ejector mode performance plots generated over the web will also be presented.

† - *Graduate Research Assistant, NASA MSFC Graduate Student Researchers Program, School of Aerospace Engineering, Student member AIAA.*
* - *Assistant Professor, School of Aerospace Engineering, Senior member AIAA.*

NOMENCLATURE

A_c	normalizing area for thrust coefficient (ft^2)
A_i	engine cross-sectional area at station *i* (ft^2)
C_t	thrust coefficient (thrust/q*A_c)
CO	carbon monoxide
CO_2	carbon dioxide
CPG	calorically perfect gas
D	hydraulic diameter
ESJ	ejector scramjet engine
ESR	ejector scram-rocket engine
Eta_{ke}	inlet kinetic energy efficiency
f	friction coefficient
H	monatomic hydrogen
H_2	hydrogen
I_{sp}	specific impulse (sec)
Kp	equilibrium constant
LOX	liquid oxygen
MR	propellant mixture ratio
O	monatomic oxygen
O_2	oxygen
OH	hydroxyl radical
P_c	rocket primary chamber pressure (psi)
P_i	partial pressure of species *i*
POST	program to optimize simulated trajectories
P_t	total pressure (psi)
phi	combustor equivalence ratio
q	freestream dynamic pressure (lb/ft^2)
RBCC	rocket based combined-cycle
SERJ	supercharged ejector ramjet engine
SESR	supercharged ejector scram-rocket engine
V	velocity
V_f	fuel injection velocity
y	axial fuel velocity component
γ	ratio of specific heats
θ_t	fuel injection angle
τ	shear stresss
ρ	density
ΔG_T	standard gibbs function change

RBCC BACKGROUND

Rocket-based combined-cycle engines are unique in that they combine the most desirable characteristics of airbreathing engines and rocket engines into a single, integrated engine. RBCC engines have the advantage of high average specific impulse (I_{sp}) in comparison to rockets, and high thrust-to-weight ratios in comparison to airbreathers.

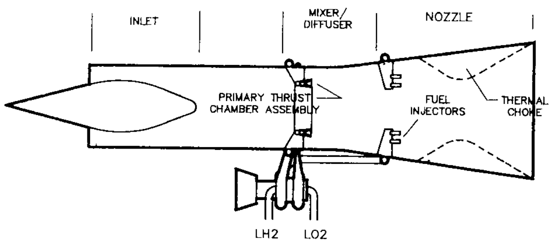

Figure 1-Supercharged Ejector Ramjet Engine (ref. 1)

The concept of combined-cycle engines has existed since the mid-60's. During this inception phase, an extensive study was conducted by the Marquardt Corporation, Lockheed-California, and the U.S. Air Force on various 'composite engine' designs, as they were formerly called [1]. This study initially analyzed 36 different variants of combined-cycle engines. At the study's conclusion, two types of RBCC engines were selected as the most interesting options — a near-term option and a far-term option. The decisions were made based on technological feasibility and resulting performance on a representative two-stage-to-orbit launch vehicle. The two final selections were the Supercharged Ejector Ramjet (SERJ) configuration (figure 1), and the more technically challenging Supersonic Combustion Ramjet with Liquid Air Cycle (ScramLACE) configuration. The SERJ engine configuration is composed of four operating modes: ejector, fan-ramjet, ramjet, and pure rocket. A derivative of the SERJ is the Supercharged Ejector Scramjet (SESJ). This configuration consists of five operating modes, the four from the SERJ and an additional scramjet mode.

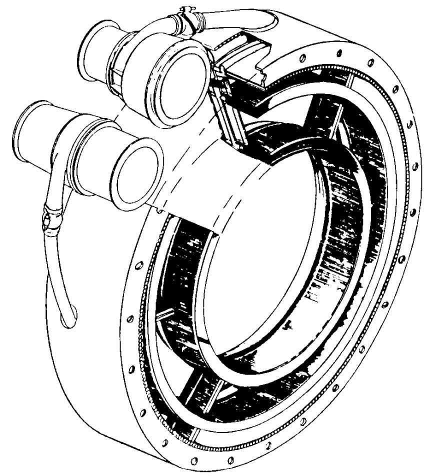

Figure 2 - Rocket Primary (ref.1)

During ascent phase, the RBCC engine initially operates in ejector mode. The ejector mode utilizes the rocket primaries (figure 2) as the main source of thrust. Entrained air from the inlet and fuel from the secondary fuel injectors is also burned in the combustor to provide additional thrust. A low-pressure ratio fan, located between the inlet and primary, may also be used. Once significant ram pressure is achieved from the surrounding air, typically occurring around Mach 2 to 3, the rocket primaries are shut off. The fan remains functioning up to about Mach 3, constituting the fan-ramjet mode. At Mach 3, the fan is removed from the flow path or perhaps windmilled in place to as high as Mach 6. The engine operates in pure ramjet mode up to around Mach 6. At Mach 6, depending upon the engine type (SESJ or SERJ), the engine will transition either to scramjet mode or directly to rocket mode. If scramjet mode is available, the engine will continue operating as an airbreather with supersonic combustion up to an optimal transition Mach number. Recent conceptual vehicle designs have suggested transition to pure rocket mode might optimally occur between Mach 10 and Mach 15. While transitioning to rocket mode, the inlet face is closed and the rocket primaries are restarted. Vacuum Isp's in the range of 410-470 seconds are typical values during rocket mode.

SCCREAM BACKGROUND

SCCREAM has the capability to model the performance of six types of RBCC engines. One of these configurations is the one identified in the Marquardt study — the supercharged ejector ramjet (SERJ). The other five are the (non-supercharged) ejector ramjet (ERJ), the ejector scramjet (ESJ), supercharged ejector scramjet (SESJ), ejector scram-rocket (ESR), and supercharged ejector scram-rocket (SESR). Additionally, SCCREAM can model pure ramjet and pure scramjet configurations.

SCCREAM operates by solving for the fluid flow properties (velocity, temperature, pressure, mass flow rate, gamma, specific heat capacity, etc.) through the various engine stations for each of the engine operating modes. Equations for conservation of mass, momentum, and energy are used. This process is often iterative at a given engine station or between a downstream and an upstream station. The flow properties are calculated using quasi-1D flow equations. Engine cross-sectional area is the only geometry variable along the stream direction. Component efficiencies are used to simulate losses of total pressure in the mixer and nozzle, and reduced enthalpy in both the rocket primary and main combustor. A started inlet is simulated by a simple total pressure recovery schedule. If the inlet is not started, SCCREAM places a normal shock in front of the cowl. Mass capture for the started inlet is determined by the flow conditions at the cowl leading edge. Mass capture for the unstarted inlet is based on the maximum allowable mass flow at the inlet throat. Thrust and I_{sp} are determined using a control volume analysis of the entering and exiting fluid momentum and the static pressures at the inlet and exit planes.

Most internal areas of the engine are determined in SCCREAM based on ratios of the inlet/cowl cross-sectional area. Default area ratios are supplied, so typically a user enters only the inlet area. The size of the rocket primary unit is primarily based on a user-entered propellant mass flow rate for the rocket primary. These two independent variables can be varied to produce an engine with a desired sea-level static thrust and secondary-to-primary mass flow ratio. In practice, however, the inlet area is often limited by overall vehicle geometry or shock-on-lip conditions. Optionally, the user can enter a desired sea-level static thrust and inlet area, and SCCREAM will iterate to determine the primary mass flow rate required.

In order to generate a POST engine table, a candidate engine's performance is evaluated over a range of altitudes and Mach numbers [2]. These Mach number and altitude ranges can be set by the user. For example, a ramjet's operational Mach numbers might be set from 2 to 5.5, with altitude ranges from 30,000 feet to 150,000 feet. Overlapping Mach numbers and altitudes between various operating modes allows POST to select optimum engine mode transition points if desired. Default Mach number and velocity ranges are provided for each mode.

Performance in pure rocket mode is determined by analyzing a high expansion ratio rocket engine operating in a vacuum. A user-entered nozzle efficiency is used to account for losses associated with the expansion of the primary exhaust through the engine and then onto the aftbody.

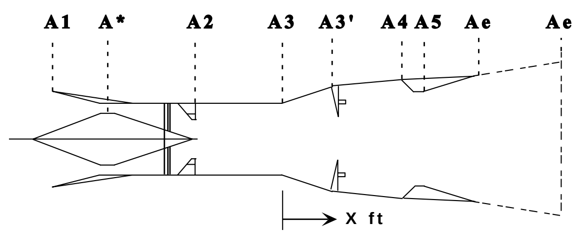

Figure 3 - Axisymetric Engine Station Locations

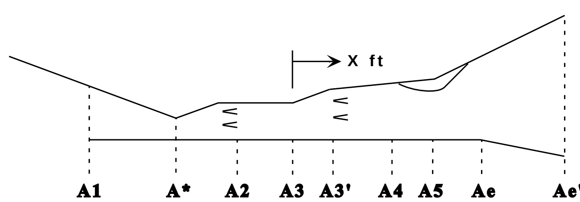

Figure 4 - 2-D Engine Station Locations

Figure 3 shows the station numbers and reference areas used by SCCREAM for an axisymetric RBCC engine configuration. Figure 4 shows station locations for a 2-D engine configuration. The 2-D engine layout is more common for vehicles with scramjet capability.

Station 1 is at the inlet plane of the engine. Freestream flow conditions at station ‘infinity’ are modified by a single shock wave to simulate any precompression effects of the vehicle forebody on the engine. The forebody shape (wedge or cone) and the forebody angle are entered by the user. Therefore the flow conditions at station 1 are typically not the same as the freestream flight conditions.

The started inlet performance is modeled by a curve fit of the total pressure (P_t) recovery for subsonic combustion and a kinetic energy efficiency (Eta_{ke}) for supersonic combustion. Both are functions of the Mach number at the inlet face. Variable geometry at the inlet throat is assumed.

Station 2 is at the location of the rocket primary. For ejector mode, station 2 to 3 is a constant area mixing process between the entrained air stream and primary exhaust.

From station 3 to 4, the fuel is injected at a specified position, velocity, angle, and equivalence ratio. A heat release profile provided by the user controls the reaction rate of the fuel. Upon exiting the combustor, the flow is passed through a converging-diverging nozzle to the exit plane of the engine (station e or e’).

For a more complete description of the flow process, the reader is referred to references 3 and 4.

SCCREAM v.5

The following is a list of the improvements made to SCCREAM that will be discussed in the following sections.

1. Improved Combustor Modeling
2. Hydrocarbon Rocket Primary Propellants
3. Hydrocarbon Afterburner Fuels
4. Scram-Rocket Mode Analysis
5. On-line Plotting Capability
6. Primary and Afterburner Phi Throttle
7. Expert and Novice Versions

Each of these improvements and the method used to implement them will be discussed in detail.

IMPROVED COMBUSTOR MODELING

A 1-D combustor model that accounts for the effects of mass addition, heat addition, friction, and variable area has been implemented in SCCREAM v.5. SCCREAM was previously limited to a constant area, frictionless combustion processes. This new model has resulted in only a minor increase in SCCREAM's total runtime.

The combustor model uses a common analysis method known as 'Influence Coefficients'. This technique allows for all the flow properties to be determined by simply solving for the change in Mach number throughout the combustor. The solution for the Mach number variation involves solving an ordinary differential equation of the form:

$$\frac{dM}{M}=C_1\left\{-\frac{dA}{A}+\frac{\gamma M^2}{2}\frac{4\,fdx}{D}+C_2\frac{dT}{T}+\left[C_3-y\gamma M^2\right]\frac{d\dot{m}}{\dot{m}}\right\} \quad (1)$$

where,

$$C_1=\frac{(1+\frac{\gamma-1}{2}M^2)}{1-M^2} \quad (1a)$$

$$C_2=\frac{\left(1+\gamma M^2\right)}{2} \quad (1b)$$

$$C_3=\left(1+\gamma M^2\right) \quad (1c)$$

The derivation of Eqn. (1) will not be presented (the authors will refer the interested reader to reference 5), but the significance of each term in the equation will be discussed.

The first term in Eqn. (1) is the impact on the Mach number due to area changes in the combustor. With the user specified area ratios and combustor lengths, this value can easily be computed. After the geometry is defined, the combustor is discretized into 1,000 axial steps.

The second term accounts for the effect of the wall shear stresses on the flow. The user provides an average skin friction coefficient, defined as:

$$f = \frac{\tau}{\frac{\rho V^2}{2}} \qquad (2)$$

where τ is a shear stress. Common values for f range from 0.001 to 0.002. The variable 'D' is the flows 'hydraulic diameter' and is the mean diameter the flow experiences at each combustor step. Note that this method assumes a circular combustor cross section.

The third term is for the heat release due to chemical reactions. The 'influence coefficient' method requires the heat release to be specified in terms of a total temperature. The procedure used for determining this will be discussed later.

The last term in Eqn. (1) is the contribution from the injected fuel's mass and velocity. This term is only nonzero at the location of the fuel injection. The 'y' term has the simple form shown in Eqn. (3).

$$y = \frac{V_f \cos(\theta_i)}{V_{stream}} \qquad (3)$$

where V_f is the velocity of the fuel, θ_i is the injection angle of the fuel, and V_{stream} is the velocity of the flow just upstream of the fuel injection position. If the fuel is injected normal to the flow, the injection angle is 90^o and the value for 'y' will be zero. If the injection angle is parallel to the flow (θ_i=0), 'y' will have its maximum value and contribution to the flow momentum. If the fuel injection occurs at any other angle, the 'y' term accounts for the axial contribution of the fuel injection.

The conditions at the entrance to the combustor are known based on the outflow conditions from the mixer section (station 2 to station 3) of the engine. To begin the analysis, the required flow properties are: the initial total enthalpy, static pressure, static temperature, Mach number, mass flow rates, and the rate of change in mass flow rate composition through the combustor. Linear profiles for the species rates of change are assumed.

The user is required to specify the friction coefficient, injection location of the fuel, start of the heat release, end of the heat release, fuel injection angle (θ_i), and fuel injection velocity (V_f). SCCREAM allows the user to specify different fuel injection and heat release parameters for subsonic and supersonic combustion cases.

Returning now to the calculation of the heat release profile. With the initial enthalpy and composition known, the total temperature can be evaluated. This is an iterative procedure which involves guessing the total temperature, evaluating each species enthalpy, mass averaging their enthalpies, and comparing the new total enthalpy with the specified initial enthalpy. Until the point of fuel injection and chemical reations start to occur, the total temperature will remain unchanged and at this value in the combustor. Thus, the 3rd term in Eqn. (1) will be zero up to the point of fuel injection.

At the location of the fuel injection, a new total temperature must be evaluated. The mass flow rate of the fuel is added to the flow composition and any enthalpy contribution from the fuel is added to the flow. The same iterative procedure performed for the initial temperature is repeated for the new mass injection total temperature.

Now, the final total temperature must be determined. This is easily accomplished since the final mass flow rates at the end of the heat release have been specified beforehand. The same procedure is then executed, using the final mass flow rates and total enthalpy, to determine the final total temperature value.

With a specified fuel injection location, starting point and ending points of heat release, a linear distribution based on the change in total temperature can be established. Note that the heat release in a real engine is most likely not a linear profile. The authors have assumed a linear profile due to its simplicity and SCCREAM can easily be modified to accommodate other profiles.

The main limitation of the influence coefficient method is due to an assumption made in its derivation. The influence coefficients assume a calorically perfect gas (CPG). Without this assumption, the relatively simple form presented above could not be arrived at . Instead of Eqn. 1, a set of partial differential equations, the Euler equations, for a 1-D flow would be arrived at. The CPG assumption typically does not hold above 2,000 R. But, by updating the specific heat and specific heat ratio at each spatial step, the usefulness of the technique has been expanded. This will be demonstrated and become evident in the subsequent verification process.

A fourth order Runge-Kutta method is used to solve Equation (1) for the variation in Mach number. The general procedure is outlined next.

All conditions at the upstream step are known beforehand. The differentials of total temperature, area, and mass flow rate are determined for the current position in the combustor. Using the Runge-Kutta solver, the local change in the Mach number is evaluated. The Mach number at the downstream point is then determined from Equation (4).

$$M_2 = M_1 + \frac{dM}{dx} dx \qquad (4)$$

A check on the value of this Mach number is performed next. A comparison against a critical Mach number is done. The critical Mach number is either 0.95 or 1.05 depending if subsonic or supersonic combustion is occuring. If the combustion is subsonic and the new Mach number exceeds the critical Mach number, or in the supersonic case, the Mach number is less than the critical Mach number, then the combustor analysis stops and a reduction in the equivalence ratio (phi) is required to prevent choking. SCCREAM automatically reduces phi and restarts the combustor analysis. If a phi reduction is not required, then the axial position, area, and total temperature are updated to the downstream position and become the upstream conditions for the next starting point.

Next, the flow composition and molecular weights are updated at the new position. A new static temperature based on the new Mach number and total temperature is then computed. With this static temperature, a new specific heat and gamma can be determined. The static pressure is then determined based on conservation of mass, using the new mass flow rate.

This entire procedure is repeated until the end of the combustor is reached, or the flow violates the critical Mach number condition. After the flow conditions at the exit of the combustor have been obtained, they are passed into the nozzle routine as entrance conditions.

Validation:

To verify the new combustor model, comparions cases were run for a scramjet combustor with the industry accepted code, SRGULL. SRGULL uses a 1-D Euler routine for its combustor analysis. This method is similar to the influence coefficient method, but without the assumption of a CPG at each local step. Similar to SCCREAM inputs, the SRGULL user can establish the engine geometry, fuel injection position, heat release profile, and combustor efficiency. One of the main differences between the SRGULL and SCCREAM model is that SRGULL uses an equilibrium chemistry routine, as opposed to SCCREAM's complete-combustion model. Properly modeling the chemistry becomes more important at higher flight Mach numbers. But, the SCCREAM model has proven adequate up to Mach 12, with engine performance numbers becoming progressively more conservative at higher Mach numbers [6].

For the test cases, a hydrogen fueled scramjet engine flying at a freestream Mach number of 6.5 and 8 were examined. The vehicle flying this engine had a 9º 2-D wedge forebody and flew along a constant dynamic pressure boundary of 2,000 psf. The combustor geometry modeled in SCCREAM and SRGULL were identical and corresponded to a ESJ engine design. Figure 5 provides the area ratio's versus axial position for the test case.

After SRGULL completed its analysis, the conditions at the entrance to its combustor were used as the entrance conditions to the combustor in SCCREAM. These conditions included mass flow rate, static pressure, static temperature, Mach number, and gamma. A constant friction coefficient of 0.0018 was specified. The fuel injection occured at an X/C value of 0.55, which also corresponded to the start of the heat release. A linear profile was established in SRGULL, which is the standard profile used in SCCREAM, and the end of the heat release was at an X/C value of 0.95. Parallel fuel injection at a velocity of 6,000 ft/s was specified for both models. At both the Mach 6.5 and 8 conditions, an equivalence ratio of 1.0 was allowed without causing a thermal choke in the combustor.

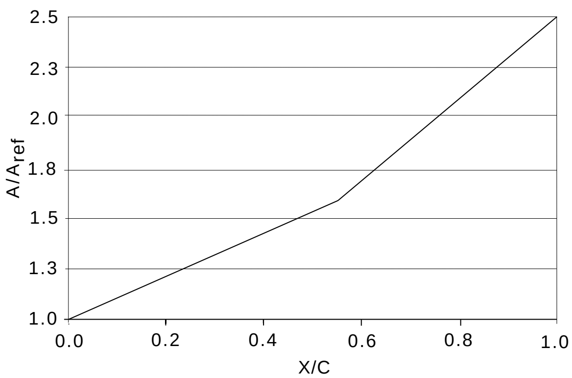

Figure 5 - Combustor Geometry

Figure 6 shows the Mach number distribution generated by SCCREAM and SRGULL for the Mach 8 flight conditions. The Mach 6.5 results were very similar and have not been included for brevity.

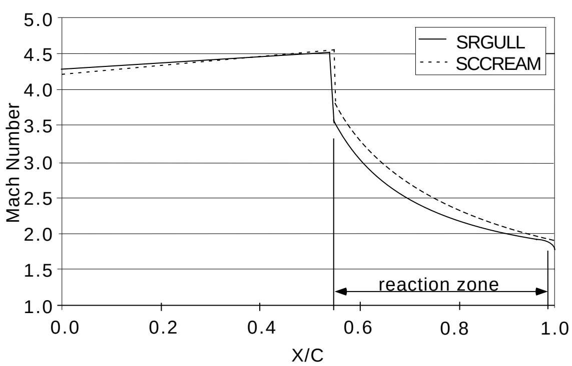

Figure 6 - Mach Number Distribution

In the non-reacting region of the combustor (up to X/C=0.55), SCCREAM and SRGULL have nearly identical profiles. At the location of the fuel injection, the sudden drop in Mach number is due to the addition of the fuel. SCCREAM appears to slightly underpredict the strength of this drop, but the effect is clearly captured by SCCREAM. The remaining portion of the combustor is the chemically reacting region. SCCREAM and SRGULL both display very similar trends and profile shapes over the entire heat release process. Differences between the two curves appear to be caused by the initial differences from the fuel injection. If the magnitude of these effects agreed, it is believed that the heat release profiles would agree almost exactly.

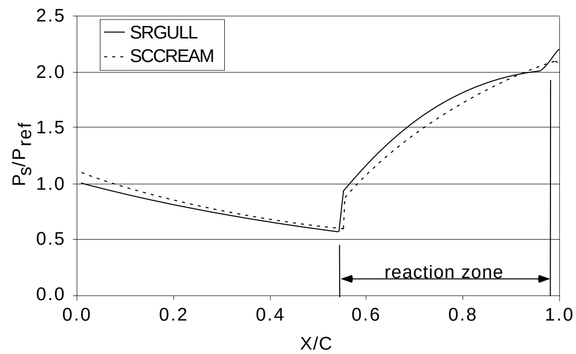

Figure 7 - Static Pressure Distribution

Figure 7 shows the static pressure distribution in the engine. Once again, excellent agreement is obtained between the two codes. The Mach 6.5 static pressure distibution displayed similar trends and will not be presented again for brevity.

HYDROCARBON ROCKET PRIMARY

Two new propellant combinations have been made available to the SCCREAM user for analysis. Either a methane (CH_4) and oxygen or a JP-5 ($C_{10}H_{19}$) and oxygen engine can be selected. These propellant combinations are in addition to the previous hydrogen (H_2) and oxygen or mono-propellant hydrogen-peroxide (H_2O_2) modeling capabilities.

To enable the analysis of these new primary subsystems in SCCREAM, a procedure similar to that used for the hydrogen and oxygen propellants was used. This procedure involves running sweeps of chamber pressure (P_c) and mixture ratio (MR) in the Chemical Equilibrium with Applications code, CEA, for each of the propellant combinations. For the methane fuel, MR's from 2 through 6, P_c's of 500 to 4,500 psi were examined. For the JP-5 fuel, MR's of 1.9 through 4.4, at P_c's of 500 to 4,500 psi were run. After each analysis by CEA, the results for each species mole fractions, chamber specific heat ratio, and chamber temperature were recorded. Using a statistical analysis software, called JMP, Response

Surface Equations (RSE's) were generated for each species concentration, chamber specific heat ratio, and chamber temperature. For improved accuracy, separate RSE's were generated for fuel rich and fuel lean mixture ratios [2,7,8].

With this model of the rocket primary combustion process, the RSE's are used to determine the mass flow rates of each species exiting the rocket primary to be combined with the flow in the mixer section during ejector mode and scram-rocket mode operation. The user is required to specify P_c, MR, and an expansion ratio. The exit conditions from the primary thruster can then be determined from isentropic 1-D flow equations. The exhaust from the primary is a contribution to the main flow's momentum in the mixer section. These same RSE's can also be used for the all-rocket mode performance analysis.

HYDROCARBON AFTERBURNER FUELS

Three new afterburner fuels have been added into SCCREAM. These fuels are methane (CH_4), JP-5 ($C_{10}H_{19}$), and JP-10 ($C_{10}H_{16}$). Previous versions of SCCREAM only provided the user with a hydrogen fuel afterburner. Addition of these fuels required adding in each of the new fuel's properties, hydrocarbon product properties, and new chemistry routines into SCCREAM.

Sensible enthalpy and specifc heats as a function of temperature for methane, JP-5, CO_2, and CO were obtained from a variety of sources and polynomial curve fits were generated. The current references for the JP-5 only included data for temperatures up to 4,000 R. If the temperatures exceed this range, values are linearly extrapolated to the required temperature. Due to the small differences between the fuels, the JP-10 fuel uses the same sensible enthalpy and specific heat curve fits as the JP-5 fuel [9].

Two different chemistry routines were required to implement the new hydrocarbon analysis. For fuel lean cases, a complete combustion with an efficiency factor can be used. For fuel rich cases, an equilibrium analysis is required.

For the fuel lean case, a simple atom balance for the reaction can be applied. Equation (5) shows the chemical reaction for the fuel lean case:

$$C_xH_y + O_2 \rightarrow CO_2 + O_2 + H_2O \qquad (5)$$

where x is the number of carbon atoms and y is the number of hydrogen atoms in the fuel chain. For the complete chemistry assumption, there will not be any CO nor minor species (O, H, OH) formed in the products. The user-specified fuel efficiency is used afterwards to compute additional concentrations of the O_2, H_2 and CO species. The efficiency is applied directly to the CO_2 and H_2O mass flow rates. The inefficency values of the mass flows are then distributed according to the reactions:

$$H_2O \rightarrow H_2 + \frac{1}{2}O_2 \qquad (6)$$

$$CO_2 \rightarrow CO + \frac{1}{2}O_2 \qquad (7)$$

For the fuel rich case, the number of product species increases and the chemical reaction under consideration becomes:

$$C_xH_y + O_2 \rightarrow CO_2 + CO + H_2 + H_2O \qquad (8)$$

Due to the presence of two C-O molecules, this reaction cannot be solved by simple atom balances. An additional equation is required and can be obtained from the water-gas shift equilibrium reaction:

$$CO_2 + H_2 \rightarrow CO + H_2O \qquad (9)$$

This reaction indicates how the carbon and oxygen molecules will be distributed between the CO and CO_2 molecules. It can be written in a more useful form in terms of the equilibrium constant, Kp, shown by Equation (10).

$$Kp = \exp\left(\frac{-\Delta G_T}{R_u T}\right) = \frac{\left(\frac{P_{CO_2}}{P^o}\right)\left(\frac{P_{H_2}}{P^o}\right)}{\left(\frac{P_{CO}}{P^o}\right)\left(\frac{P_{H_2O}}{P^o}\right)} \qquad (10)$$

Kp is significant because it is only a function of temperature. Because this reaction is bimolecular, the pressure terms on the right hand side will cancel out, leaving only mole fractions. This eliminates the pressure dependance of the reaction and greatly simplifies the problem. With this setup, we now only require knowledge of the static temperature at the exit of the combustor. After careful consideration, it was decided that using the total temperature at the entrance to the combustor provided a good estimate of the static temperature at the exit of the combustor. Ideally, this would be an iterative procedure consisting of making a guess, performing the combustor analysis, then using the new static temperature for the new Kp value. It can be shown that at the elevated combustion temperatures, Kp is only mildly affected by the temperature, thus the iterative process is unnecessary [10].

With the CO_2 and CO species concentrations determined, the concentration of the H_2O molecule can found and any remaining H-atoms are in the form of diatomic hydrogen. Similar to the fuel-lean case, the efficiency is applied directly to the CO_2 and H_2O concentrations.

It should be mentioned that for ejector mode and scram-rocket mode calculations, any hydrocarbon species generated by the rocket primary, with the exception of O_2, are not considered in the afterburner chemistry analysis. This includes the species CO_2, CO, H_2O, OH, O and H. For all cases, any disassociation of nitrogen (N_2) that might be occuring at elevated temperatures is not considered.

Validation:

In order to validate the new hydrocarbon analysis capability, a sample engine configuration was established and analyzed during ramjet and scramjet operation. SCCREAM performance numbers (thrust and Isp) were generated and compared with results from the Ramjet Propulsion Analysis code, RJPA, for both the methane and JP-5 fuels [11]. RJPA was selected instead of SRGULL because it is a more common analysis tool to use in the conceptual design environment. SRGULL is more commonly used in the preliminary design stages of engine development and requires significantly more set-up time.

The comparison engine configuration was similar to the engine used for a Georgia Tech vehicle design called 'Stargazer'. Figure 8 provides an external view of the Stargazer vehicle.

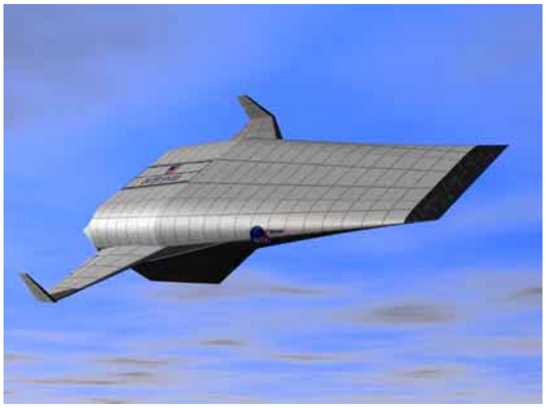

Figure 8 - Stargazer TSTO Bantam Concept

The Stargazer vehicle is a two stage concept designed to deliver a small, Bantam class payload (300 lbs) to a 200 nmi. circular orbit. The first stage of Stargazer is a RBCC ESJ booster. The second stage is an expendable, LOX-RP liquid rocket. The booster stage is unpiloted and has horizontal take off and landing capability. For the hydrogen version, the booster accelerates up to Mach 3 in ejector mode, then transitions to ramjet mode until Mach 6. At Mach 6, the engines operate as scramjets until Mach 10. After Mach 10, the inlets are closed-down and the rocket primaries are re-ignited. The booster vehicle then accelerates to Mach 15 and separates with the second stage at a dynamic pressure of 1 psf. The booster then performs a turnaround maneuver and cruises back to the launch site under ramjet power at Mach 3.5.

Stargazer has a 7^o 2-D wedge forebody. Table 1 provides the engine geometry, fuel injector parameters, and efficiency factors used for the comparative analysis.

The RJPA model was set up with an identical engine geometry, friction coefficient, injection angle, mass capture, and equivalence ratio. For ramjet operation, the normal shock model was utilized in

RJPA. Because of the extreme sensitivity of the total pressure recovery to the diffuser exit area, it was difficult to exactly match the inlet total pressure recoveries with those used by SCCREAM. In scramjet mode, RJPA allows specification of the inlet efficiency (Eta_{ke}) aand the same value used by SCCREAM was used in RJPA.

Table - 1 Stargazer ESJ Engine Data

inlet area, A_1	20.0 ft^2
mixer area, A_3	10.0 ft^2
combustor break, $A_{3'}$	12.0 ft^2
combustor exit, A_4	16.8 ft^2
maximum exit area, $A_{e'}$	65.0 ft^2
combustor efficiency, η_c	95.0%
nozzle efficiency, η_{nozz}	98.5%
friction coefficient, f	0.001
fuel temperature, T_f	500.0 R
fuel injection velocity, V_f	2,000 ft/s
fuel injection angle, θ_i	0.0 deg

Figure 9a shows the thrust coefficient (C_t) versus Mach number results for the methane fuel cases. The physical cowl area (A_1) of the engine was used to non-dimensionalize the thrust to obtain the C_t. At Mach 3, the maximum equivalence ratio allowed is 0.3. At Mach 4, 5 and 6 an equivalence ratio of 1.0 can be obtained. In ramjet mode, SCCREAM appears to underpredict the thrust levels produced by RJPA. Most of this discrepancy is being attributed to the difference in the total pressure recovery. For scramjet mode operation, SCCREAM and RJPA match very closely. All scramjet mode points are at an equivalence ratio of 1.0. The maximum difference in the thrust coefficient is only 5%, which occurs at a Mach number of 8.

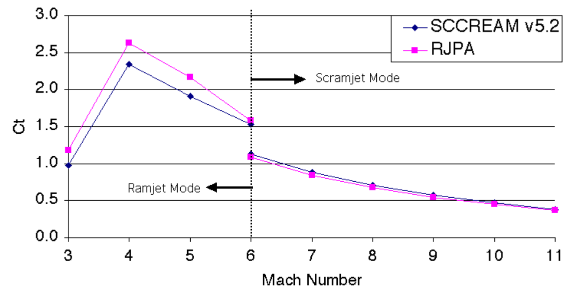

Figure 9a - Methane Thrust Coefficient

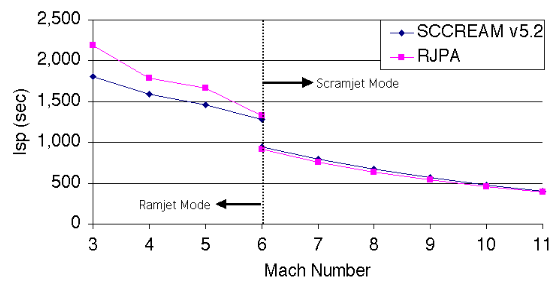

Figure 9b - Methane Specific Impulse

Figure 9b shows the Isp versus Mach number results for the methane case. The same trends shown in the thrust coefficient plots are also displayed in the specific impulse charts.

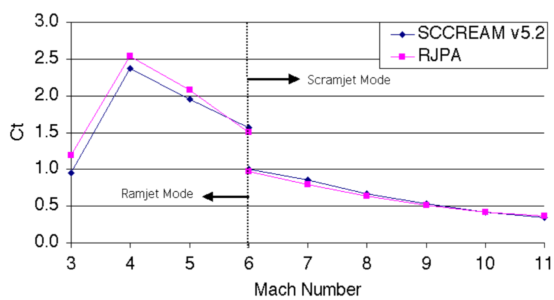

Figure 10a - JP-5 Thrust Coefficient

Figure 10a shows the results for the same engine geometry and flight conditions, but with JP-5 fuel. As with the methane fuel, SCCREAM and RJPA both display similar trends. The maximum difference in scramjet mode thrust is 7% and occurs at Mach 7. This difference is being attributed to the heat release profile. RJPA does not perform a marching solution through the combustor, thus the effects of fuel injector placement are not accounted for. These differences in the thrust values are within the tolerances of different heat release profiles and injector locations.

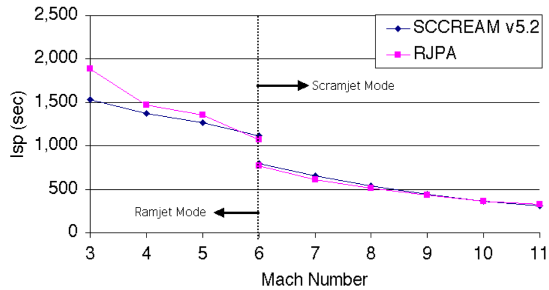

Figure 10b - JP-5 Specific Impulse

Figure 10b shows the results for the specific impulse for the JP-5 fuel case. Similar trends to those in the thrust coefficient plots are also shown.

SCRAM-ROCKET MODE ANALYSIS

The additional operating mode of a rocket-augmented scramjet (ESR or SESR engine) has been added. At high Mach numbers, the scramjet thrust can become low enough that the vehicle cannot accelerate properly. The previous solution to this problem was to shut down the scramjet, quickly transition the engine to all-rocket mode, and make an abrupt exit from the atmosphere. In addition to wasting the oxygen still available in the atmosphere, this pullup maneuver can significantly increase the wing loading of the vehicle. The preferred method is to augment the scramjet mode with the rocket primary and gradually transition to full-throttle all-rocket mode. This will allow the vehicle to continue accelerating while still using the free oxygen contained in the air. This results in a higher specific impulse than the all-rocket mode and significantly increased thrust compared to the scramjet mode.

Figure 11a provides the results from SCCREAM's scram-rocket mode analysis case compared with the scramjet mode performance and all-rocket mode performance. The scram-rocket results presented are for the case of the rocket primary being ramped up incrementally, starting at Mach 9 and having full-throttle at Mach 12. A common technique used in trajectory simulation is to simply ramp down the scramjet thrust and ramp up the rocket thrust in order to model a gradual switch between the modes. The results of using this technique are also plotted. It can be seen that the simple averaging technique does not capture all of the flow dynamics created by the rocket primary exhausting inside of the scramjet engine. It is also interesting to note that the scram-rocket mode provides 15% more thrust than the all-rocket mode at Mach 12.

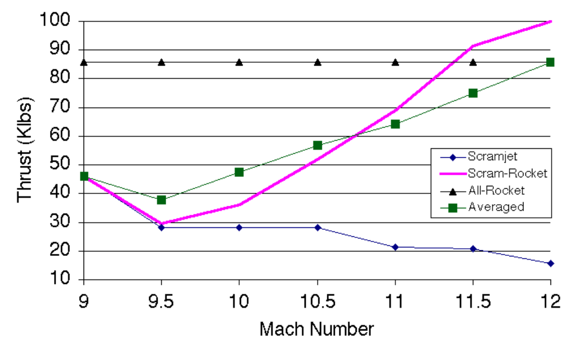

Figure 11a - Scram-Rocket Mode Thrust

Figure 11b shows the Isp values for the scram-rocket mode performance. Also included are the all-rocket mode Isp adjusted for the altitude, scramjet mode Isp values, and the 'averaged' Isp values.

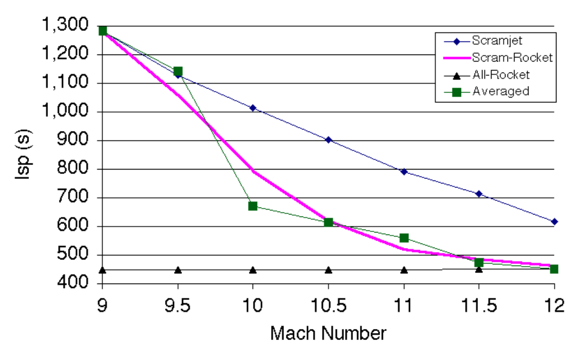

Figure 11b - Scram-Rocket Mode Isp

ON-LINE PLOTTING CAPABILITY

The SCCREAM web site at:

http://titan.cad.gatech.edu/~jebradfo

is continuously being improved. The latest capabilities in dynamic HTML and JavaScript are being used to increase the capabilities and improve the work environment for the SCCREAM user. Previously, the output returned by SCCREAM was limited to a POST engine deck and tables consisting of detailed results at each design point. In addition to these, the web interface now returns performance plots for ejector mode engine performance.

To allow plotting of the ejector mode data, additional operations must be performed by the CGI script. These operations involve the use of freeware plotting packages and a graphics interpreter. This software must be accessible to the web CGI script in order to generate the plots.

After the SCCREAM executable has successfully finished analyzing the engine's performance, a PERL script executes a 'C' program that parses through the POST engine deck and creates a set of arrays that can be read by the plotting package.

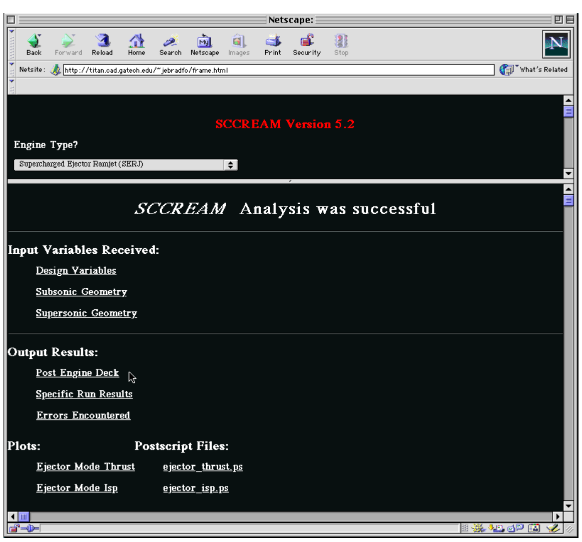

Figure 12 - SCCREAM v.5 Output Page

After the 'C' script has finished parsing through the POST deck, the freeware plotting package called 'GNUPLOT' is executed. A script with commands for GNUPLOT is piped into the executable from a command line call executed by the PERL script. GNUPLOT then reads the arrays generated by the 'C' script and creates two different 2-D plots of thrust and Isp versus Mach number at every altitude analyzed. The first plot set are color postscript files and the second are black and white postscript files (PS). The color postscript file is later converted to a color JPEG image using the freeware program 'ghostscript'. This JPEG image can then be displayed to the user over the web. The PS file can also be downloaded by the user and provides a crisp, black and white image for the user to send to their local printer [12,13].

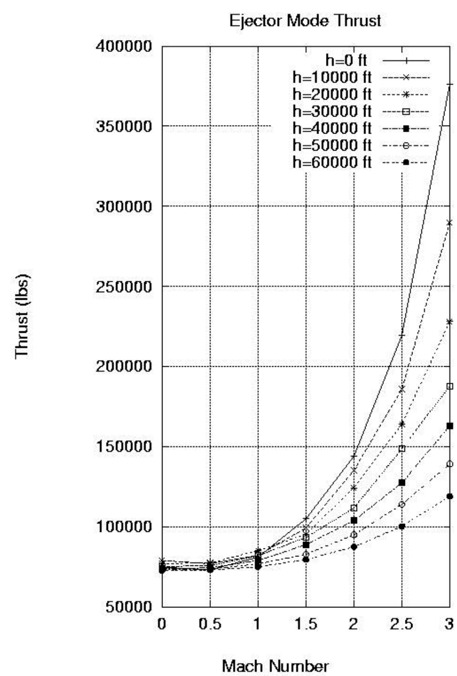

Figure 13a - Ejector Mode Thrust Plot from Web

Figure 12 is a snapshot from the SCCREAM web site showing the various links to the engine data after a successful analysis. Figures 13a and 13b are JPEG images taken directly off the SCCREAM web page. They were generated for a Supercharged Ejector Ramjet (SERJ) engine with a sea-level static thrust of 75,000 lbs and fan pressure ratio of 1.2. The rocket primary used LOX/JP-5 propellants and the afterburner used JP-5.

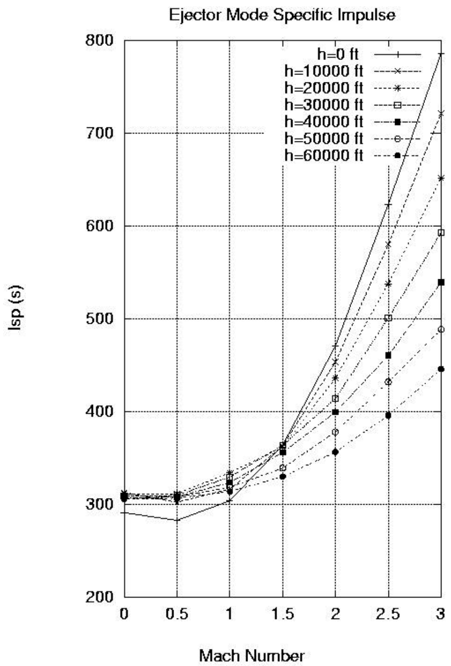

Figure 13b - Ejector Mode Isp Plot from Web

The new plotting capability allows the designer to quickly assess their engine's performance and provides an easy method of comparing different engines.

ROCKET PRIMARY AND AFTERBURNER PHI THROTTLE

SCCREAM now has the ability to create POST decks for throttled engines. This will allow the trajectory analyst more control over optimizing the flight path of the vehicle. The rocket primary is throttled in ejector mode and the remaining modes throttle the afterburner equivalence ratio. Figure 14 provides a sample POST deck for a throttled engine's ramjet mode Isp. The throtle is represented by the variable 'genv5'. For example, at an altitude of 50,000 feet, a speed of Mach 3.0, and a phi throttle of 0.667, the engine produces an Isp of 1612.06 seconds.

```
c Ramjet Mode Isp Values
l$tab table=5hisp3t,3,5hgenv5,
      4hmach,5hgdalt,4,4,5,27*1,
50000,
   3.0,
          1,      1413.32,
      0.667,      1612.06,
      0.333,      1845.88,
          0,      0,
   4.0,
          1,      1460.47,
      0.667,      1572.53,
      0.333,      1738.73,
          0,      0,
  5.0,
          1,      1338.87,
      0.667,      1372.43,
      0.333,      1221.65,
```

Figure 14 - Sample Throttled POST Deck

EXPERT AND NOVICE VERSIONS

Two different web interfaces for SCCREAM are now available to the user, an 'expert' mode and a 'novice' mode. The latter is a 'scaled-down' interface which should facilitate and encourage the use of SCCREAM in classroom environments.

The original SCCREAM interface is now the 'expert' interface. This allows the user full access to all of SCCREAM's modeling options and design variables. For the engineer who is new to the RBCC engine design process, the 'novice' interface is recommended. This version has significantly fewer number of design variables than the 'expert' mode. All second-order engine parameters like fuel injection velocity, heat release profile, station efficiencies, etc.. have been set to nominal values. The novice user is first encouraged to understand the basic principles of designing an engine. This includes thrust level to inlet area matching at sea-level static conditions, engine geometry effects on subsonic and supersonic combustion, and propellant/fuel type analysis.

CONCLUSIONS

SCCREAM's analysis capability has been significantly expanded since version 4. As the tool continues to develop, it is expected that its usage will become even more commonplace. Over the past year, web access to SCCREAM has increased significantly. A number of universities have used SCCREAM in student engine and vehicle design competitions and are now hosting their own SCCREAM web sites. Web statistics indicate the Georgia Tech SCCREAM site has been accessed by users from 8 different countries. In the past year, SCCREAM has analyzed over 1,500 different engine designs (over 500,000 flight conditions)

Among the conclusions drawn from this paper are the following:

1. The new combustor model has been shown to compare very well with the preliminary design tool, SRGULL's modeling capability. This increased level of fidelity has not significantly increased the run-time for SCCREAM. Hundred's of flight conditions can still be analyzed in under 2 minutes.

2. SCCREAM is now capable of modeling 4 different rocket primary propellant combinations and 3 different afterburner fuels. These new capabilities have been validated with the industry accepted code RJPA. SCCREAM is flexible enough to model any combination of these propellants and fuels.

3. The new scram-rocket capability will allow for improved modeling accuracy of the scramjet to all-rocket mode transition. It also provides a wider flight envelope for the trajectory optimization process.

4. The web-based plotting capability will enable SCCREAM users to quickly and conveniently view and interpret their engines performance.

FUTURE WORK

SCCREAM will continue to be improved with the intent of increasing modeling accuracy and capabilities without sacrificing speed, ease of use, and flexibility. Among many near-term improvements being considered for version 6.0 are the following:

1. Improving the inlet pressure recovery model by adding geometry dependent shock system analysis. This will replace the current curve fit models. A method for incorporating a normal shock model for subsonic combustion modes has been devised but not implemented. A technique for supersonic combustion cases is still under development.

2. A propane rocket primary subsystem and afterburner fuel option will be added. The procedure for implementing this will be consistent with current methods of analysis for hydrocarbon fuels.

3. The post-SCCREAM analysis plotting capability will be expanded to include all engine modes, as well as 3-D contour plots.

4. A method will be established for determining angle of attack effects. This is a fairly simple procedure for the wedge configuration, but there does not appear to be a quick solution for conical flows at an angle of attack. Once generated, these effects will be added to the POST deck for incorporation into the trajectory analysis.

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