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Improvements and Enhancements to SCCREAM, A Conceptual RBCC Engine Analysis Tool

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ABSTRACT

A rocket based combined-cycle engine analysis tool suitable for use in the conceptual design environment has recently been established. While this tool was being used in the design environment, new analysis capabilities were desired and areas for improvement were noted.

This paper will detail the recent improvements made to the conceptual design tool, SCCREAM, and present the results generated by the added capabilities. The improvements range from an additional engine analysis mode, alternate propellant combinations, and a new user-interface which enables remote execution.

The improvements and added capabilities to SCCREAM will be discussed and the program methodology will be examined in detail when appropriate. Results generated by SCCREAM's new scramjet analysis mode are then shown to compare very well with an industry standard code, RJPA. Engine performance generated by SCCREAM for a single stage to orbit launch vehicle are then compared with historical airbreathing engine performance data, and other industry common analysis codes.

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NOMENCLATURE

A_c	normalizing area for thrust coefficient (ft^2)
A_i	engine cross-sectional area at station *i* (ft^2)
Ar	argon
C_p	constant pressure specific heat (BTU/slg-R°)
C_t	thrust coefficient (thrust/q*A_c)
H_2O_2	hydrogen peroxide
H	monatomic hydrogen
H_2	hydrogen
I_{sp}	specific impulse (sec)
K	kelvin
LH2	liquid hydrogen
LOX	liquid oxygen
MR	propellant mixture ratio
N_2	nitrogen
O	monatomic oxygen
O_2	oxygen
OH	hydroxyl radical
P_c	chamber pressure (psi)
P_t	total pressure (psi)
phi	combustor equivalence ratio
q	freestream dynamic pressure (lb/ft^2)
V_r	radial velocity component
V_θ	normal velocity component
γ	ratio of specific heats
θ	ray angle from cone centerline

RBCC BACKGROUND

Rocket Based Combined-Cycle (RBCC) represents a new approach for providing routine access to space. By integrating the elements of rocketry and air-breathing systems into a single unit, RBCC tries to exploit the best qualities of each. The rocket primary is used for providing the high level of thrust required at takeoff conditions and for acceleration until ramjet takeover speeds can be obtained. Once ramjet operation is feasible, the rocket primary is shut off to conserve fuel. The airbreathing modes of ramjet and scramjet are then used to accelerate the vehicle through the portions of the atmosphere where free oxygen is available. As the vehicle climbs and increases its speed, a point will be reached at which the ramjet or scramjet is no longer providing enough thrust to sufficiently accelerate the vehicle. For single stage to orbit (SSTO) configurations, it is at this point that the rocket primary is re-ignited and the vehicle proceeds directly to orbit.

RBCC is not a new concept. Originating in the 1960's, a variety of basic concepts were developed considerably under a joint effort by the Marquardt Corporation, U.S. Air Force, and Lockheed[1]. Due to budget constraints at the time and technical challenges required for full implementation, RBCC quickly fell to the sidelines, and the less complex rocket engine received full attention for space applications.

During the 1980's, significant gains were made in the area of airbreathing propulsion. The National Aerospace Plane program, or NASP, made major technological gains for airbreathing systems. NASP identified the major difficulties associated with this form of propulsion and many new technologies in the areas of thermal protection, inlet design, and supersonic combustion were enabled. Despite the technology advances, the unbelievable and overwhelming task of airbreathing to speeds above Mach 15 prevented a feasible vehicle design from being obtained.

It has been only recently that interest has been renewed in RBCC systems. By merging two previously independent systems, RBCC can offer a number of advantages for launch vehicle designers. In terms of engine performance, RBCC offers higher trajectory averaged specific impulse (I_{sp}) than pure rocket engines, and higher engine thrust-to-weight ratios than pure airbreathing engines. But, these gains come at the expense of a higher vehicle dry weight and increased vehicle complexity. The real advantage from RBCC is in the high flight rates and mission flexibility that these engines enable. RBCC is suitable for missions that include: earth-to-orbit, pop-up trajectory maneuvers, and high speed point-to-point missions. RBCC also promises increased loiter and abort options. These capabilities will be required on future space transportation systems.

A number of very attractive vehicle concepts for future launch systems have already been designed[2]. Many of the most promising of these concepts utilize RBCC propulsion, and the feasibility of these systems is almost unquestioned. The primary challenge now is in designing an economically viable system. With total program development costs ranging in the billions of dollars, robust designs that ensure success are mandatory.

RBCC propulsion appears to have a very promising future, and may provide the key to affordable, routine, and safe access to space.

PREVIOUS RESEARCH

Engineers in a conceptual RBCC launch vehicle design environment needed to be able to assess engine performance at each point in the ascent trajectory. That is, for a given altitude, flight velocity, and engine operating mode, what thrust and I_{sp} are produced by the engine? This data is typically used in a trajectory optimization code to determine a minimum fuel flight path to orbit.

Due to computing speed limitations, the required engine data is commonly generated off-line for a range of expected altitudes and flight speeds. The resultant database is formatted into a tabular form. Data is interpolated from the tables as needed by the trajectory optimization code.

The current engine analysis tool, SCCREAM, is a descendant of tools generated under earlier research efforts[3]. SCCREAM (Simulated Combined-Cycle Rocket Engine Analysis Module), is an object-oriented code written in C++. The code executes on a UNIX workstation, runs a full range of flight conditions and engine modes in under 60 seconds, and will output a properly formatted POST[4] engine table. SCCREAM is not intended to be a high-fidelity propulsion tool suitable for analyzing a particular RBCC engine concept in great detail, although its results compare very well with those generated from more detailed codes. It was created to be a conceptual design tool capable of quickly generating a large number of reasonably accurate engine performance data points in support of early launch vehicle design studies.

SCCREAM OVERVIEW

SCCREAM has the capability to model the performance of four types of RBCC engines. One is the configuration identified in the Marquardt study —the supercharged ejector ramjet (SERJ). The other three are the (non-supercharged) ejector ramjet (ERJ), the ejector scramjet (ESJ), and the supercharged ejector scramjet (SESJ). 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. The inlet is simulated by a simple total pressure recovery schedule. 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 in SCCREAM are determined based on ratios to 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. 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.

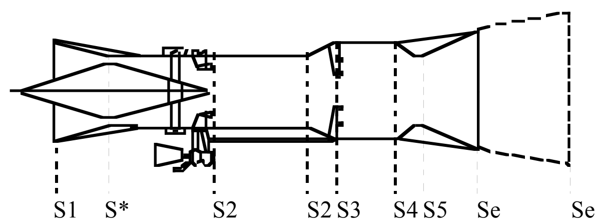

Figure 1 - Axisymetric Engine Station Locations

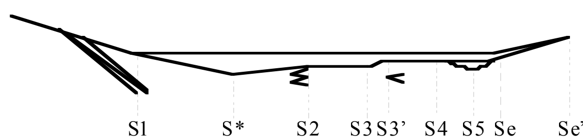

Figure 2 - 2-D Engine Station Locations

Figure 1 shows the station numbers and reference locations for by SCCREAM for an axisymetric RBCC engine configuration. Figure 2 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 inlet performance is modeled by a curve fit of the total pressure recovery and is a function 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 and scramjet fuel injectors. 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 3' an isentropic expansion of the flow is performed. This is generally beneficial for ramjet performance, but tends to penalize the scramjet performance.

From station 3' to 4, the hydrogen fuel is injected at a specified equivalence ratio and allowed to burn. 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 Reference 3.

IMPROVEMENTS

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

1. Scramjet analysis capability
2. Rocket primary combustion
3. Rocket primary propellants
4. Detailed forebody analysis
5. New POST output deck format
6. Remote operation

Some of the improvements have already been mentioned while discussing the general operation of the code. Each will now be discussed in detail.

Scramjet Analysis

As stated earlier, the previous version of SCCREAM lacked a scramjet mode analysis capability. Results from an earlier study by Shaughnessey[5] were hard-wired into SCCREAM for this mode. The scramjet capability is undoubtedly the most significant and important improvement made.

Modeling of scramjet performance involved allowing a supersonic flow to pass completely through the engine without choking in the inlet throat, combustor, and nozzle sections. The conservation equations for mass, momentum, and energy were employed in a similar manner to that from the subsonic flow (ejector, fan-ram, and ramjet modes) cases. By careful arrangement of the iteration routines, the supersonic solution which satisfies the 3 conservation equations can always be obtained.

The entire mass flow at the inlet face is always ingested by the engine. The flow at station 1 is passed through the inlet and oblique shock system (not actually modeled in detail). A curve fit for the total pressure recovery of a supersonic inlet, based on the Mach number at station 1 replaces the subsonic inlet curve fit. Figure 3 shows the subsonic and the new supersonic pressure recovery schedules.

The conditions at the location of the rocket-primary (station 2) are then determined. This is a simple iteration procedure and as long as the area blockage from the rocket-primary is not too large, then a supersonic Mach number at station 2 can be obtained. If the area downstream of the inlet is too small, a common occurrence for RBCC configurations with oversized primaries, the downstream primary blockage will choke the flow to subsonic conditions. For these cases, a solution is not obtained.

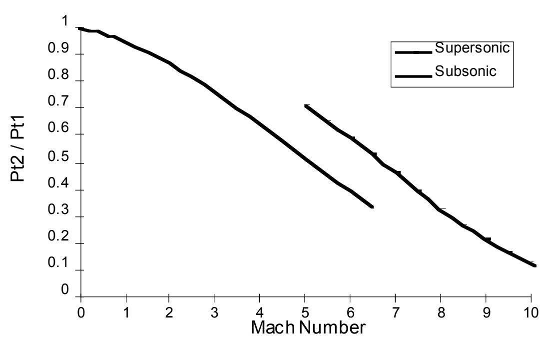

Figure 3 - Inlet Total Pressure Recovery

The hydrogen fuel is injected and mixed from station 2 to station 3, without any reaction occurring (no heat addition). This is done to simulate injecting the fuel further upstream, as often required for supersonic combustion to allow adequate mixing. The added fuel changes the molecular weight and specific heat of the flow. This slightly affects the static conditions at station 3. A total pressure loss is simulated in the mixer section by defining an efficiency factor.

When solving for the static conditions at station 3, a new iteration procedure is required. Recall that for the subsonic flow cases, the assumption was that the static temperature is close to the total temperature. The mixture specific heat capacity was then calculated using the total temperature. This allowed for a much simpler iteration routine involving Mach number, which can easily be bounded between Mach 0 and Mach 1. For supersonic flow, the assumption of the static temperature being close to the total temperature is poor. A new routine has been devised that now includes the specific heat in the determination of the static conditions.

The static temperature at station 3 is iterated upon instead of the Mach number. This creates some problems because of the difficulty in setting upper and lower bounds on the temperature that will always ensure a supersonic solution is obtained. The exact problems encountered will be discussed later.

With an assumed static temperature and known flow composition, the mixture specific heat can be obtained. JANNAF based curve fits of the specific heat for each species a function of temperature is used by SCCREAM. A mass averaging technique is then used to determine the specific heat of the mixture.

Once the specific heat is obtained, the specific heat ratio can then be easily calculated since the molecular weight is known. The total enthalpy of the flow at station 3 is the same as that a station 2, thus the total temperature can be obtained dividing the total enthalpy by the specific heat value.

The known quantities are now static temperature, total temperature, and specific heat ratio. From these, the Mach number at 3 can be obtained using the conservation of energy equation. From the definition of Mach number, the flow's velocity can then be obtained. The continuity equation, or conservation of mass, is then used to determine the static pressure at station 3.

It is now necessary to obtain a new value for the static temperature to confirm the guessed value. This new temperature is obtained from the momentum equation. It is assumed that the added fuel has no contribution to the momentum balance.

The new and guessed temperatures are then compared and a new estimate for the static temperature, based upon a bisection routine, is determined. This process is repeated until convergence is obtained.

As previously mentioned, convergence problems can be encountered from iterating on the static temperature. If the guess is to high, the flow can have a subsonic Mach number at station 3. But, this condition can also result if too much fuel is added at station 2. A series of checks is used to either adjust the guess for the temperature or reduce the amount of fuel being added.

After a solution at station 3 is reached, the flow is isentropically expanded to the area at station 3'. This process will accelerate the flow since it is supersonic and the area is increasing.

From station 3' to station 4, the fuel that was added in at station 2 is now burned. The combustion process is modeled as a frictionless, one-dimensional heat addition process. A routine similar to that used from station 2 to 3 is applied again. At station 4, a minimum Mach number at or above sonic conditions can be set, with the SCCREAM default being Mach 1.15. If thermal choking occurs, or the minimum Mach number constraint is violated, the amount of fuel added (based on user defined phi) is automatically reduced, and the analysis restarts at station 2. Complete combustion is assumed, with the combustor efficiency accounting for the unburned fuel and resulting oxygen content. Species accounted for in the combustion process are: N_2, H_2O, Ar, O_2, and H_2.

After station 4, the flow is expanded out the diverging portion of the nozzle to the exit plane of the engine (station e) or aftbody of the vehicle (station e'), depending upon the current flight altitude. Since the flow is supersonic, there is not a converging section in the nozzle. The flow composition from the combustor is frozen, and the specific heats are again included in the iteration procedure to account for the decreasing static temperature from the accelerating flow.

It should also be noted that SCCREAM can also be used to model a pure ramjet or pure scramjet engine now. These are non-RBCC engines configurations that do not have an ejector-mode nor the accompanying blockage at station 2 in the engine.

Rocket Primary

Previously, the user had large number of input parameters that had to be defined in order to properly model and size the rocket primary. These parameters included the total temperature, molecular weight, specific heat ratio, expansion ratio, and chamber pressure. SCCREAM was able to accurately determine the primary nozzle exit area and product exhaust velocity, but only after the user had over-defined the primary. Once in the engine, the flow was then assumed to be composed of 100% H_2O and the user-defined value for the molecular weight was overridden and set to 18.0, corresponding to a pure steam exhaust. Thus, even after defining all these inputs, the rocket primary could still only be modeled at stoichiometric conditions upon entering the main engine.

To eliminate this discrepancy and relieve the user of the extraneous input parameters, Response Surface Equation's (RSE's) were used to model the chamber temperature and exhaust product mole fractions. RSE's model complex systems with simple algebraic equations. These equations can yield very accurate results for non-discrete models, as well as save valuable computation time.

In all, 8 RSE's were generated as a function of the chamber pressure and mixture ratio. The first two equations were for the total temperature and specific heat ratio(γ). The remaining 6 were used for the mole fractions of: H_2, O_2, H_2O, O, H, and OH.

The well established Chemical Equilibrium and Applications program, or CEA[6], from the NASA Lewis Research Center was used for determining the equilibrium composition in the rocket chamber. For the analysis performed by CEA, the rocket propellants (oxygen and hydrogen) were both assumed to be in gaseous form at 298 K. The input parameters, chamber pressure and mixture ratio, were varied from 500 to 3,000 psia and from 4 to 12 respectively. A total of 64 different cases were analyzed.

After all of the runs were completed, a statistical analysis program , JMP[7], was then used for setting up the RSE's. The general form of each RSE generated is:

$$X = \alpha * P_c + \beta * P_c^{\,2} + \chi * MR * P_c + \delta * MR + \varepsilon * MR^2 \quad (1)$$

where P_c is the chamber pressure, MR is the primary mixture ratio, and α through ε are constants. A residual analysis of the RSE fits show excellent correspondence with the results from CEA.

With the mole fractions now known, the molecular weight of the mixture can be determined. The flow composition is frozen and then expanded to match the user-defined expansion ratio. Basic rocket analysis equations are used for solving for the throat area, exit pressure, and exit velocity.

SCCREAM was then modified to track all of the primary exhaust products throughout the rest of the engine. In doing so, operation of a non-stoichiometric rocket primary is enabled.

Primary Propellants

An additional rocket primary propellant, hydrogen peroxide (H_2O_2), has been added. Concentrations of 85%, 90%, and 98% H_2O_2 can be selected and modeled. The non-H_2O_2 percentage in the concentrations is pure water. The user is simply required to select the desired concentration, then enter the chamber pressure and expansion ratio for the primary subsystem.

Hydrogen peroxide is a mono-propellant that reacts when brought into contact with a catalyst like platinum or copper. For a given concentration, the decomposition temperature is fixed, thus the expected temperatures for the 3 concentrations are hard-wired into SCCREAM. The decomposition of H_2O_2 results in a mixture composed of 43% O_2 and 57% H_2O by weight, not including any initial H_2O present. Benefits of H_2O_2 are the design simplicity resulting from having only a single working fluid, as well as a lower combustion temperature. The lower combustion temperature allows for increased chamber pressures. Typical values for P_c are from 500-5000 psi. These benefits come at the cost of a lower specific impulse and exhaust velocity.

SCCREAM will analyze the performance and size the rocket primary for the H_2O_2 configurations. Industry data has shown that 100% decomposition is nearly obtainable, so a primary combustion efficiency is not used for these cases. The O_2 and H_2O exhaust products are then tracked through the mixer and into the combustor. The excess O_2 from the primary is added to the oxygen content of the air stream. The total oxygen mass flow is then used with the equivalence ratio to determine the amount of fuel added in the combustor.

Forebody Analysis

SCCREAM allows the user to define either a conical or 2-D wedge shaped forebody to account for compression effects. In SCCREAM version 1.0, both the cone and wedge shapes used closed form solutions for solving for the flow properties behind the bow shock. For conical flow, this closed form solution will accurately predict the properties behind the shock, but not behind the shock at the surface of the vehicle and at the cowl lip. To obtain a more accurate estimate of the mass flow at the inlet, a more rigorous analysis is now performed.

For determining the properties behind the bow shock of a cone, a system of 3 ordinary differential equations must be solved. They are shown here in their more familiar (spherical coordinates) form:

$$V_\theta = \frac{\mathrm{d}V_r}{\mathrm{d}\theta} \qquad (2)$$

$$\frac{dV_\theta}{d\theta} = [\frac{a^2}{(V_\theta^2 - a^2)}](2 * V_r + V_\theta * \cot\theta - \frac{V_r V_\theta^2}{a^2}) \qquad (3)$$

$$\frac{dp}{d\theta} = (\frac{-\rho V_\theta a^2}{(V_\theta^2 - a^2)}) * (V_r + V_\theta \cot\theta) \qquad (4)$$

where a is the speed of sound, θ is the ray angle, V_r is the radial velocity component, V_θ is the normal velocity component perpendicular to the radial component, p is the static pressure, and ρ is the density of the flow.

The reverse procedure of guessing a shock angle, as recommended by Anderson[8], is implemented to solve these equations.

Additional information about the inlet is required from the user (these input values are not necessary for a wedge shaped forebody). These new inputs are the length from the nose of the vehicle to the inlet lip and the height of the inlet.

When equations (2)-(4) are solved, the flow field behind the bow shock is completely defined. A streamline that intersects the cowl lip can be determined using the additional input parameters. The mass flux is then determined along this streamline and averaged with the mass flux at the vehicle's surface. This value is then used as the mass flow rate seen across the entire inlet at station 1.

Output Deck

The static pressures inside an airbreathing engine can be substantial and will significantly effect the weight of an engine. The trajectory flown by the vehicle will have the strongest influence on the maximum internal pressures that will be experienced by the engine. For freestream dynamic pressures (q) greater than 1500 psf, ramjet mode static pressures in excess of 200 psi can easily develop as the flight Mach number is increased. This can significantly increase the weight of an engine, and this information needs to be supplied to the engine weight model.

Figure 4 shows the maximum static pressures experienced by an Ejector Scramjet configuration for a typical constant-q, single stage RBCC vehicle. Note that from Mach 4 to 5.5, the pressure increases very rapidly, especially for the q=2000 psf trajectory. At Mach 5.5, the q=1500 path has a maximum pressure of only 220 psi, while the q=2000 path experiences over 300 psi. These effects are indeed significant and must be accounted for in the overall vehicle design.

To allow for tracking of these engine pressures, a new table has been included in the POST engine deck produced by SCCREAM. This table contains the maximum static pressure experienced by the engine at every flight condition. This information can easily be monitored during the trajectory, and can be passed to an engine weight estimation code (WATES[9]) during each iteration while closing a design. Alternatively, a maximum static pressure limit can be set in the trajectory model. POST can be constrained not to exceed this value for the static pressure over the course of the trajectory.

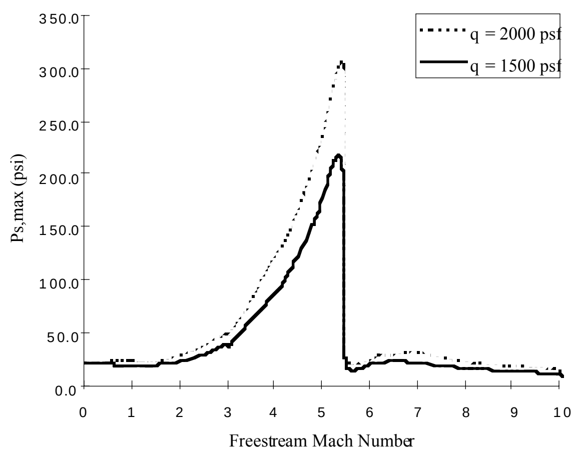

Figure 4 - Maximum Internal Static Pressures

Remote Operation

In the interest of allowing easy access and operation of SCCREAM, a web based interface has been created. This interface allows for execution and retrieval of the results from SCCREAM over the web from any computing platform. The user must simply have access to an Internet browser (Netscape, Internet Explorer, etc.). The web interface also allows for any user to easily access the most current version of SCCREAM without the hassle of obtaining and installing the newest version. Currently, access to SCCREAM is unrestricted. The web address for SCCREAM is:

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

In addition to remote operation, the new interface allows for easy error checking before program execution. Hyper-links for each variable are set up to provide a brief description of each input parameter and give typical ranges. Sample engine configurations for a variety of RBCC vehicles have also been included on the page.

The web interface is composed of three different programming languages. They are the common Hyper Text Markup Language (HTML), JavaScript, and Practical Extraction Report Language (PERL).

The HTML portion utilizes the form 'post' method for transferring data to the machine hosting the SCCREAM executable. The 'post' method is preferable over the 'get' method when transferring more than one piece of information. The web page itself consists of radio buttons, pull-down menus, and text fields for the SCCREAM input parameters. This allows for easy configuration changes and updating of the engine model. Figure 5 provides a partial screen shot of the user interface for Version 4.0 of SCCREAM.

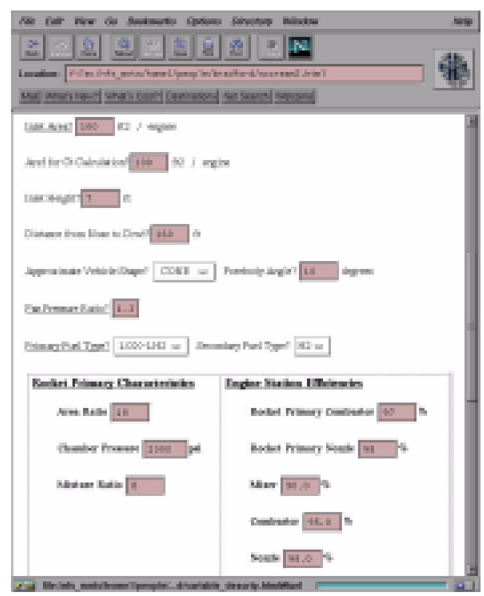

Figure 5 - Web-based user interface

The JavaScript routines perform error and range checking of the user inputs. This helps limit the possibility of errors being generated when SCCREAM executes. For example, if the user accidentally puts in a nozzle efficiency greater than 100%, a JavaScript warning message will be displayed. This message will identify the name of the variable with the infeasible input value and provide the allowable ranges for the particular variable. The JavaScript also creates a more dynamic page, with default input values automatically changing based upon a user's selections. As an example, if a non-supercharging RBCC engine (no fan) is selected, the fan pressure ratio automatically changes to 1.0, for no total pressure rise. If the user selects the pure-ramjet option, all input fields associated with the rocket primary subsystem are eliminated.

Once all of the input parameters have been checked and verified by the JavaScript, an estimate of the total run time required is displayed. The web form is then processed by execution of a Common Gateway Interface (CGI) script. This script is located on the server for the SCCREAM host, and is written in PERL. This PERL script opens and writes to the 6 text based input files, runs SCCREAM, and then displays the results back to the user's web browser. It should be noted that the original text input files are still in place and the SCCREAM source code has not been altered to be compatible with the web interface. Therefore, SCCREAM can still be executed on a stand-alone platform that does not have Internet access.

After execution of SCCREAM is completed, the user can simply download the results by selecting the hyper-links to the main output file and POST deck. The browser 'Save As' option will retrieve the results and place them in the user's local directory.

RESULTS

Comparison with RJPA

The Ramjet Performance Analysis Code[10], RJPA, was developed at Johns Hopkins University in the mid-1960's. The Fortran based code uses a one-dimensional integral analysis approach and is applicable to a wide variety of airbreathing and rocket propulsion concepts. The combustor uses the NOTS equilibrium code for determining the chemical composition of the flow. Frozen and equilibrium flow analysis options can be selected.

The RJPA engine model is divided into 4 main components: the inlet, diffuser, combustor, and nozzle sections.

For comparison runs with SCCREAM, only scramjet performance was analyzed for Mach numbers from 6 to 12. A generic scramjet engine configuration with moderate internal area contraction and exit flow expansion was selected.

For establishing the inlet flow conditions, the static conditions for temperature, velocity, and pressure behind the bow shock were specified for each case. These values were obtained from SCCREAM for a conical forebody with a half-angle of 9.2°. The physical area of the inlet at the cowl was 51 ft^2.

The diffuser section consisted of defining the exit area, total pressure recovery, and initial guesses for the specific heat ratio. The exit area from the diffuser corresponded with the area at station 3' in SCCREAM, and was set to a value of 33 ft^2. The total pressure recovery was made to correspond to the value used by SCCREAM, at each flight condition. Heat losses in the diffuser were ignored.

For the combustor model, a constant area process was desired, so the exit area from the combustor was 33 ft^2. Skin friction and heat transfer in the combustor were neglected. The equivalence ratio and initial guesses for the static pressure at the exit plane were also defined in RJPA. For cases below Mach 7.25, the equivalence ratio had to be reduced in order to prevent choking due to the heat addition in the combustor. If the specified phi is too high in RJPA, a solution cannot be obtained. For these same cases, SCCREAM automatically throttled back the fuel flow rate from the maximum value defined by the user. The phi determined by SCCREAM provided starting points for determining an allowable phi in RJPA. It should be noted that the allowable fuel flow rate from SCCREAM was slightly higher than the value allowed by RJPA. To ensure a fair comparison, SCCREAM was run again with the same phi used by RJPA.

For the nozzle expansion, an efficiency of 98% and an exit area of 204 ft^2 was defined. A frozen-to-equilibrium nozzle flow ratio of 0.667 was also used for determining the thrust and I_{sp} values. RJPA performs the nozzle analysis for both frozen and equilibrium flow. The frozen flow case should have lower thrust and I_{sp}, when compared to the equilibrium case. Real nozzle performance is somewhere in between these to bounds, with kinetic models suggesting it is closer to the frozen flow results. By defining a frozen-to-equilibrium ratio of 2/3, RJPA computes a 'real' flow performance by averaging 2/3 of the frozen flow results with 1/3 of the equilibrium flow results. The performance results presented are for the 'real' flow case.

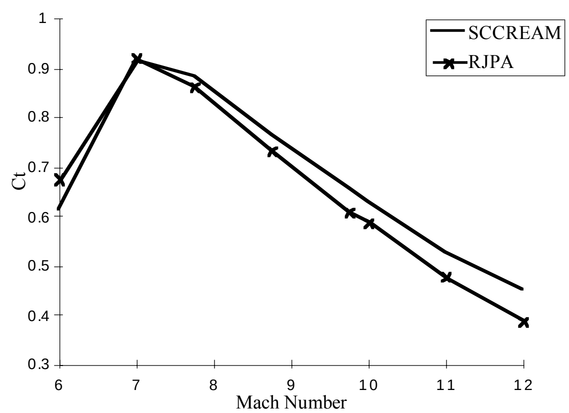

Figure 6 - C_t versus Mach Number

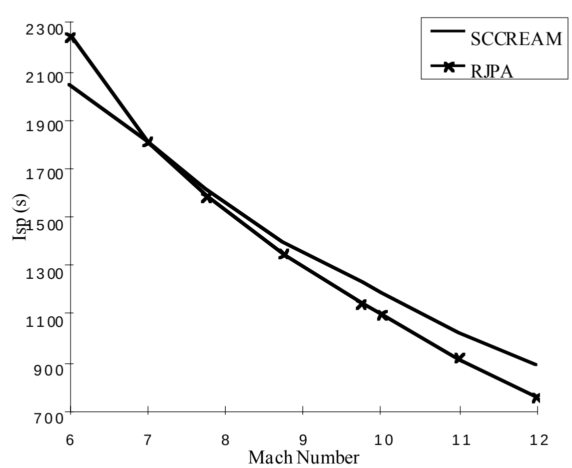

Figure 7 - I_{sp} versus Mach Number

Figure 6 provides the comparative results for the thrust coefficient versus freestream Mach number. The cowl area of 51 ft^2 was used to normalize the thrust coefficient. The dynamic pressure for most cases was approximately 2000 psf.

In the Mach number range of 7 to 10, SCCREAM and RJPA match very well. At the lower Mach numbers, it appears SCCREAM underpredicts the thrust level predicted by RJPA. This is currently being attributed to SCCREAM not modeling the pre-combustion static pressure rise (the PSPCI term in RJPA) from the shock train. This pressure rise results in different flow conditions at the start of the combustion process, which in turn effect the flow conditions exiting the combustor. Table 1 provides more detailed information on the static conditions at this low Mach number condition. Notice from the table the static pressure and temperature differences exiting the combustor. These differences diminish at the Mach 8.75 and Mach 10 conditions, where the effect of the shock train pressure rise also diminishes. This lends support as to the theory of why the differences are occurring, but determining the exact mechanism will require further investigation.

At Mach numbers above 10, the differences between RJPA and SCCREAM appear to be slowly increasing. As the Mach number and energy of the flow increases, the exact composition of the flow becomes more important. Of particular consequence is the fact the SCCREAM does not account for the hydroxyl species (OH). The presence of the hydroxyl molecule will effect the molecular weight and specific heat of the flow. These in turn affect the static conditions. Since this is not modeled by SCCREAM, a higher thrust value than RJPA could result at increased Mach numbers due to different static conditions at the exit plane.

Figure 7 provides the I_{sp} versus Mach number. As expected based on the thrust coefficient trends, SCCREAM slightly underpredicts the I_{sp} predicted by RJPA at the lower, reduced phi, Mach numbers. From Mach 7 to 10, very good correspondence between the two codes is displayed again. Above Mach 10, SCCREAM has a higher I_{sp} in a similar manner as the thrust profile.

Table 1. Flow Property Comparison for SCCREAM and RJPA

		Mach 6		**Mach 8.75**		**Mach 10**	
	Flow Property	*SCCREAM*	*RJPA*	*SCCREAM*	*RJPA*	*SCCREAM*	*RJPA*
Diffuser Exit	Mach	3.74	4.11	5.03	5.24	5.27	5.56
	Ps (psi)	3.54	3.70	2.83	2.59	2.39	2.14
	Ts (R)	605.1	722.5	768.8	1004.0	927.3	1188.60
Combustor Exit	Mach	1.21	1.09	1.96	1.87	2.36	2.31
	Ps (psi)	25.16	36.47	17.68	19.09	11.94	12.36
	Ts (R)	3195.4	4009.3	4713.9	4906.3	4792.2	4847.20
	Gamma	1.281	1.267	1.245	1.250	1.245	1.25
	Mol. Wgt	26.45	26.55	24.31	23.81	24.31	23.76
Nozzle Exit	Mach	3.17	3.12	3.49	3.43	3.84	3.80
	Ps (psi)	1.10	1.41	1.23	1.25	0.91	0.92
	Ts (R)	1643.7	1933.4	2831.1	2776.8	2908.2	2813.30
	Gamma	1.328	1.315	1.270	1.282	1.272	1.28

Comparison with Other Codes

The *Hyperion* concept's ejector scramjet (ESJ) engine performance has been reanalyzed using the current SCCREAM (Version 4.0) model. *Hyperion* is a single stage to orbit vehicle that flies on a constant q boundary of 2,000 psf in scramjet mode up to Mach 10. The vehicle is design to carry 20,000 lbs to low earth orbit from Kennedy Space Center (KSC) in Florida. The forebody is a conical shape with a half-angle of 9.2°. The reader is encouraged to obtain reference 3 for more details on the *Hyperion* concept.

As previously documented, RAMSCRAM[3,11] data has been generated based upon a similar Hyperion engine geometry and flight path. It should be noted that a ramjet to scramjet transition Mach number of 6 was used for the RAMSCRAM data, but *Hyperion* now transitions at Mach 5.5.

SRGUL was used by Shaughnessey[5] to generate ramjet and scramjet performance for a vehicle with a 5° half-cone angle for NASA-Langley. These results are for a non-RBCC engine with a different engine geometry and inlet efficiency.

The RJPA results presented here are for the conditions previously stated in the direct comparison cases. The engine geometry is very similar to *Hyperion's* engine design.

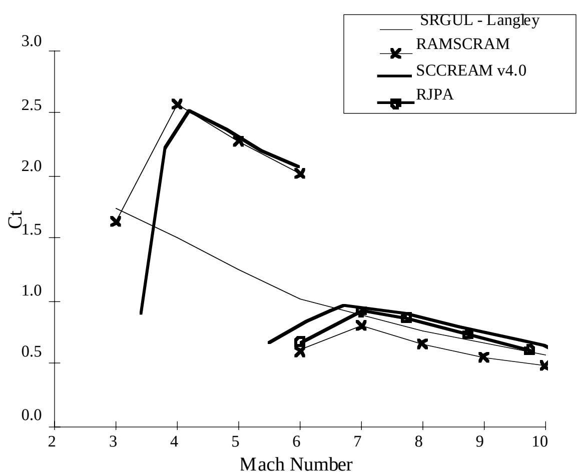

Figure 8 - C_t versus Mach Number(group 1)

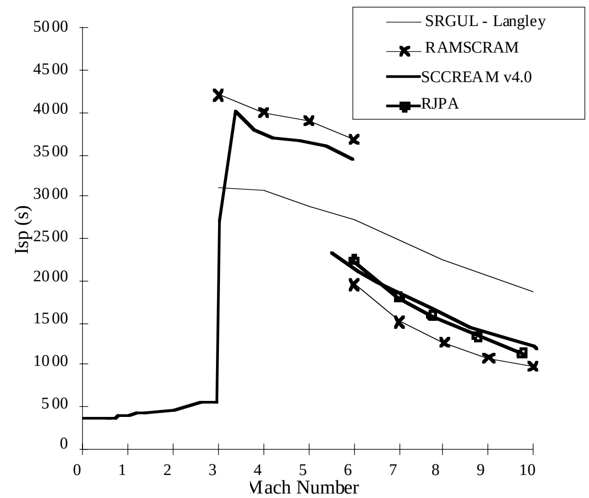

Figure 9 - I_{sp} versus Mach Number(group 1)

Figure 8 shows the thrust coefficient comparisons for the 4 codes SCCREAM, RJPA, RAMSCRAM, and SRGUL. It can be seen that SCCREAM and RAMSCRAM match very well for the ramjet portion of the trajectory. SCCREAM appears to accurately predict an equivalent drop in thrust from transitioning from subsonic to supersonic combustion. Note that an instantaneous switch from subsonic to supersonic flow is modeled here, but a real engine would likely have a much smoother transition period. In scramjet mode, RJPA and SCCREAM agree very well, as previously shown. RAMSCRAM appears to have less thrust than SCCREAM and RJPA in scramjet mode, but still displays similar trends.

The SRGUL data fits very well with all three codes at Mach numbers greater than 7. But, as Mach number decreases below Mach 7, SRGUL's thrust coefficient continues to increase while the rest are exhibiting a decrease. It is known that SCCREAM, RJPA, and RAMSCRAM have lower thrust at these Mach numbers due to throttling of the equivalence ratio. The need to throttle phi to prevent choking the flow is largely dependent on the engine geometry and inlet efficiency. It is also known that the SRGUL engine flowpath allows a phi=1 at these Mach numbers, which accounts for the increasing thrust level. Designing for the phi=1 scramjet condition can come at the expense of performance in other modes. This would not have been a consideration for the designer of a pure scramjet configuration.

Figure 9 shows the I_{sp} profiles for the 4 codes. Once again, SCCREAM and RAMSCRAM match well for ramjet mode performance. SCCREAM and RJPA match almost exactly in scramjet mode, and RAMSCRAM is displaying similar trends again. The SRGUL I_{sp} profile does not coincide with any of the codes. It should be re-iterated that this is not the same flow path design and engine configuration. The data does provide an interesting reference for comparing RBCC performance with an engine designed for only ramjet/scramjet operation.

Comparison with Historical Data

Data from the early Marquardt studies for ejector ramjet and ejector scramjet configurations has been obtained. Results from this study are commonly referred to as NAS7-377 data. This data is for a launch vehicle with an 8° half-angle wedge forebody, flying on a constant q boundary of 1500 psf.

In 1988, the Astronautics Corporation[12] performed a study for the United States Air Force. The vehicle used ejector scramjet engines and had a 10° half-angle cone. But, the data obtained and presented here are results for a 6° half-angle wedge.

Figure 10 shows the thrust coefficient profile generated by SCCREAM and compared with historical data. In the early stages of ramjet mode, the large increase in the thrust coefficient by SCCREAM can be attributed to the increasing phi in the combustor. As the flight speed increases, the maximum phi of 1 is quickly obtained. The thrust coefficient matches well with the NAS7-377 ejector ramjet predictions for the remainder of subsonic operation. SCCREAM and the trends from the Astronautics data appear to agree well in scramjet mode. Due to the differences in forebody angles, inlet efficiency, and internal geometry, it can not be expected that SCCREAM will exactly match these predictions.

Figure 11 shows the I_{sp} comparisons of SCCREAM with the historical data. It appears that agreement over the most of the trajectory is excellent. But, it does not appear that the NAS7-377 and the Astronautics data have a change in performance while transitioning from subsonic to supersonic combustion. This is curious as the recent analysis from RJPA and RAMSCRAM both display the same drop in performance being predicted here by SCCREAM. A possible explanation is that a smooth transition was modeled between ramjet and scramjet operation.

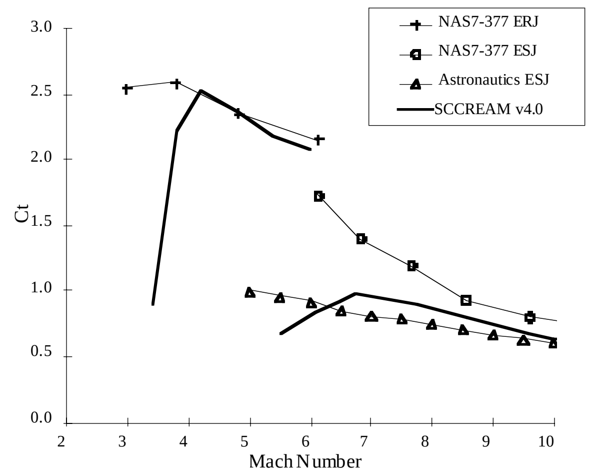

Figure 10 - C_t versus Mach Number(group 2)

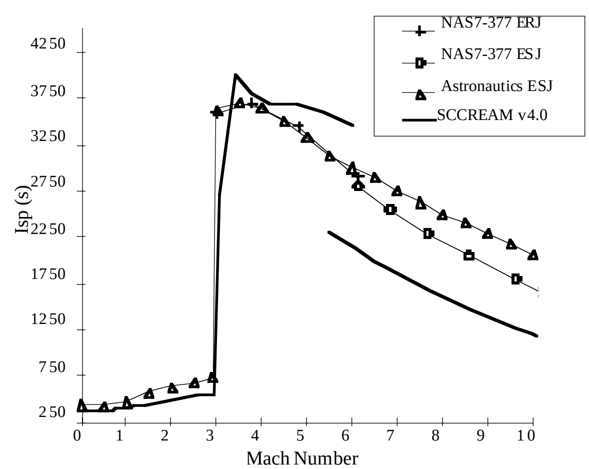

Figure 11 - I_{sp} versus Mach Number(group 2)

CONCLUSIONS

Significant improvements have been made to SCCREAM since its inception. The current version 4.0 has retained its execution speed, while at the same time improving its accuracy and capability.

Among the conclusions drawn in this paper are the following:

1. A scramjet performance model suitable for a conceptual design environment has been established. The accuracy of its results has been confirmed through direct comparison with the industry standard code, RJPA.

2. The required number of inputs for defining the rocket primary flow has been reduced from 5 to 3. This was accomplished at the same time as greatly improving the accuracy of the primary flow and increasing its modeling ability.

3. The first fuel trade study capability has been enabled by addition of a hydrogen peroxide rocket primary. This primary can be operated at 3 different initial concentrations of 85%, 90%, and 98%.

4. Valuable static pressure information has been added to the trajectory output deck. This will allow the designer to more accurately perform trades and model a vehicle's trajectory. The new data can easily be incorporated into an engine weight estimation model.

5. A web based user interface has been established. This interface readily allows remote execution, reduces the possibility of input errors, and eliminates the need for updating software by the remote user.

FUTURE WORK

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

1. Addition of a combustor model that will allow for a non-constant area and account for friction and heat loss effects. This work has currently begun.

2. Allow for specifying multiple compression ramps on the forebody surface. This will be implemented for both conical and wedge configurations.

3. Creation of the additional operating mode known as 'scram-rocket'. This mode occurs near the end of scramjet operation, while transitioning to the all-rocket mode. It has the potential of maintaining adequate thrust through use of the rocket primary, while still utilizing the small amount of oxygen in the atmosphere to increase specific impulse.

4. Establish a method 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.

5. Provide on-line data plotting using the web-based interface. This will allow the user to quickly assess their engine's performance.

6. Addition of a hydrocarbon primary and secondary fuel-injector analysis capability. Hydrocarbon fuels have been identified as promising candidates for RBCC missile applications.

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