ATR/VEHICLE INTEGRATION

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Abstract

The Air TurboRamjet (ATR) possesses design, performance and operation characteristics of the turbojet (TJ), ramjet and rocket in a single air breathing propulsion system. The basic design tradeoff of ATR design is specific thrust and specific impulse. There are many possible variations to this basic cycle, but this research has concentrated on the solid fuel gas generator driven ATR. Because overall vehicle performance is more important than engine only operation, the evaluation of the ATR as a propulsion system must be done at the vehicle level. The results of this study show that an 800 lbm solid rocket motor (SRM) powered vehicle achieves a range of about 30 miles in about 100 seconds. A 750 lbm ATR powered vehicle attains a range of about 50 miles in 192 seconds. A 525 lbm TJ powered vehicle will travel nearly 50 miles in 290 seconds. This means the ATR will provide nearly double the range of a SRM powered vehicle with roughly double the flight time. Compared to the TJ powered vehicle, the ATR achieves the same range in about two thirds of the TJ flight time. The weight difference between the ATR and TJ systems is the penalty paid for the ATR's shorter flight time.

Symbols/Abbreviations

Aexit	Nozzle Exit Area, square inches
ATR	Air TurboRamjet
Cpair	Specific Heat of Air at Compressor Entrance
Cpt	Specific Heat of Turbine Drive Gas, BTU/(lbm-deg R)
η_c	Combustor Efficiency
η_t	Turbine Efficiency
Fs	Specific Thrust, lbf-sec/lbm
γ_a	Ratio of Specific Heats of Air
γ_c	Ratio of Specific Heats of Combustor Gas
γ_t	Ratio of Specific Heats of Turbine Drive Gas
Gc	Gravity Constant [32.174 (lbm-ft)/(lbf-sec^2)]
GG	Gas Generator
GLOW	Gross Lift Off Weight, lbm
HV	Heating Value, BTU/lbm
Isp	Specific Impulse, lbf-sec/lbm
lbf	pounds force
lbm	pounds mass
MR	Mixture Ratio (fuel/air)
MRs	Stoichiometric Mixture Ratio
MW	Molecular Weight, lbm/lbMole
Pa	Ambient Pressure, psia
Pc	Combustor Total Pressure, psia
Pex	Nozzle Exit Static Pressure, psia
PRc	Compressor Pressure Ratio
PRt	Turbine Pressure Ratio
R	Universal Gas Constant [1545 lbf-ft/(lbMole-deg R)]
SFGG	Solid Fuel Gas Generator
SRM	Solid Rocket Motor
Tc	Adiabatic Flame Temp, deg R
TJ	Turbojet
Tti	Turbine Total Inlet Temp, deg R
Tt2	Compressor Air Total Temp, degR
Va	Air Velocity, feet/second
Wf	Fuel Mass Flow Rate, lbm/second

ATR General Description

The ATR possesses design, performance and operation characteristics of the TJ, ramjet and rocket engine in a single air breathing propulsion system. It is illustrated schematically in Figure 1 below.

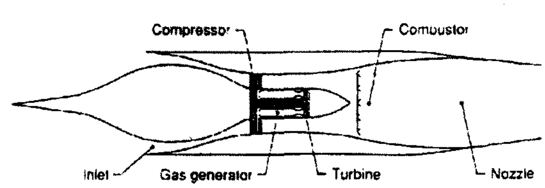

Figure 1. ATR Schematic

As shown in the figure, the turbine drive gas is supplied from a GG which operates independently of the air flow through the engine. This effectively eliminates the turbine inlet temperature as an ATR operating constraint. In fact, the turbine can be driven by almost any gas that will flow through the turbine nozzle and blade passages. The ATR can generate net positive thrust from sea level static conditions to Mach 5 to 6 flight conditions without a booster propulsion system. The ATR has an inherently higher maximum operating speed and altitude than a turbine temperature limited TJ because 1) the compressor rather than the turbine is the temperature limiting component of the ATR and 2) the TJ turbine temperature is normally higher than the compressor temperature. The ATR compressor provides a higher combustor pressure than that available from ram pressure alone. Since the minimum combustor pressure drives the maximum altitude at which the combustor can sustain combustion, the maximum operating altitude of the ATR is inherently higher than that of a ramjet.

SFGG ATR Description

The SFGG ATR is identical to the general case ATR except that the propellant is stored as a solid fuel or propellant in the GG. This also means that once the ATR is started it can be throttled up or down but cannot easily be shut down completely.

The cycle illustrated in Figure 1 can be called the basic ATR cycle. Although there are many possible variations to this basic cycle, this research has concentrated on the SFGG driven ATR. Most of the test articles built to date have demonstrated operation and throttability of the ATR at static conditions. However, vehicle performance parameters like flight time and range are the figures of merit with which the ATR must be evaluated in comparison to other propulsion systems.

Specific Impulse vs. Specific Thrust

The basic tradeoff of ATR design is specific thrust, Fs, and specific impulse, Isp. Isp, the ratio of thrust produced to fuel flow rate, is the most important propulsion system performance parameter. Isp should always be as high as possible, while meeting any other system constraints such as size, weight and cost. Fs relates the performance to the size of the engine. As Fs increases, this means that the air flow, for a given thrust level, is dropping. As air flow rate drops, the compressor and inlet size also drop. Combustor size must also be considered. Combustor size can be reduced by using a higher pressure ratio compressor which, because it drives combustor pressure, will generally require a smaller combustor. A higher PRc will also reduce compressor and inlet size since, all other engine parameters being the same, a higher PRc requires a higher MR. As will be seen, a higher MR tends to increase Fs. Also, size has a very significant effect on vehicle performance since engine size affects the aerodynamic characteristics of the vehicle.

The relationship between Fs and Isp can be seen in the following equations which define 1) MR of the ATR at all operating conditions (design and off-design), 2) Isp as a function of MR and 3) Fs as a function of MR.

The first equation is defined by a power balance between the compressor and turbine. The equation assumes first law characterizations of both the compressor and turbine modified by an efficiency for each component. The resulting expression is:

$$MR = \frac{T_{T_2}\, C_{P_{air}} \left(PR_c^{\frac{\gamma_a - 1}{\gamma_a}} - 1 \right)}{\eta_c\, \eta_t\, T_{t_i}\, C_{p_t} \left(1 - \left(\frac{1}{PR_t} \right)^{\frac{\gamma_t - 1}{\gamma_t}} \right)} \quad (1)$$

Note that the HV of the fuel does not appear in this expression. This is because power extraction from the turbine drive gas is not a function of its heat release potential when combusted with air.

Related expressions for both Fs and Isp can also be defined analytically in terms of MR. In this way, turbomachinery characteristics such as PRc, η_c, PRt and η_t can be related directly to Fs and Isp.

The second equation defines Isp in terms of MR.

$$I_{sp} = \frac{MR+1}{MR} \sqrt{\frac{2R\gamma_c \eta_c T_c}{G_c(\gamma_c - 1) MW_c} \left(1 - \left(\frac{P_{ex}}{P_c} \right)^{\frac{\gamma_c - 1}{\gamma_c}} \right)} + \frac{P_{ex} - P_a}{W_f} A_{exit} - \frac{V_a}{MR\, G_c}$$

Here, The first two terms in this equation represents the Isp that would result if the combustor gas was expanded isentropically to Pex. Note that the combustor gas temperature Tc is multiplied by η_c to define the actual gas bulk temperature. The second term represents the increase in Isp due to the pressure difference which acts over Aexit. The last term represents the Isp loss due to ram drag. This Isp loss increases as MR drops since this means there is more air and hence more ram drag.

If all parameters in this equation are held constant, except for MR, a plot of Isp vs MR shows that as MR drops, Isp increases and as MR increases, Isp drops. This trend generally holds even though Tc varies with MR for any given fuel. Unlike rocket engines, ATR Isp continues to increase as MR decreases. In other words, the lower the ATR MR, the higher the theoretically possible Isp. The maximum Isp is ultimately limited by Pc which is dependent on PRc. Rocket engine Isp, on the other hand, maximizes at a definite combustion chamber MR, usually close to the stoichiometric MR.

Essentially then, within limits imposed by either the flight condition and/or components, as MR increases, more fuel is required to produce a given thrust thus reducing Isp. Conversely, as MR drops, less fuel is required to produce the required thrust and thus Isp rises. Therefore, in general, any component design parameter change that reduces MR will enable a higher Isp and vice versa.

The third equation for Fs as a function of MR is easily derived from the Isp expression. This equation is shown below:

$$Fs = (MR+1)\sqrt{\frac{2R\gamma_c\eta_c T_c}{G_c(\gamma_c-1)MW_c}\left(1-\left(\frac{P_{ex}}{P_c}\right)^{\frac{\gamma_c-1}{\gamma_c}}\right)} + \frac{MR}{W_f}(P_{ex} - P_a)\,A_{exit} - \frac{V_a}{G_c} \quad (3)$$

As this equation shows, Fs increases with MR. The first two terms relate to the thrust generated by expansion of the combustor gas from the main nozzle. The last term is the Fs loss due to ram drag .The net effect of increasing design point MR, then, is to increase Fs.

The effect of MR on both Isp and Fs is illustrated in the following graph which uses equations [2] and [3] above. It shows both Isp and Fs as a function of MR for a higher HV (approx 11000 BTU/lbm) solid propellant and a lower HV (6000 BTU/lbm) propellant. This plot clearly shows the tradeoff between Isp and Fs.

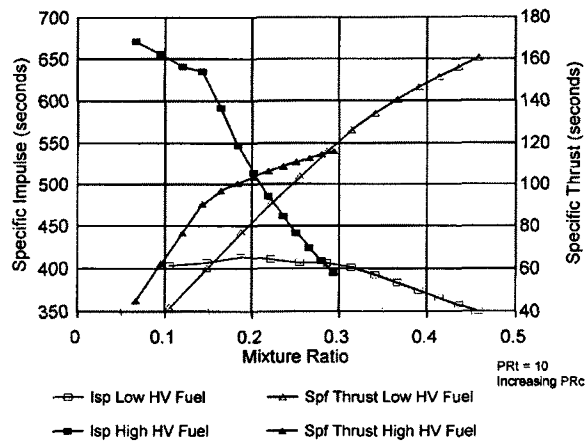

Figure 2. Isp & Thrust vs MR

For this figure, PRt was held constant and PRc was varied. The higher HV propellant provides superior Isp and Fs at lower MR values. However, as the MR increases, the lower HV propellant will provide the superior Isp and Fs. It should be remembered that each propellant has a feasible range of operating MR that is established by the turbomachinery characteristics and gas properties as defined in equation [1].

In both equations, and as illustrated in the figure, Isp and Fs are continuous functions of MR. Curves of Isp and Fs vs MR are smooth, regardless of whether the ATR is operating fuel or air rich. The lower MR limit is established by the turbomachinery characteristics. As either PRc drops or PRt increases, MR will drop. Assuming 100% efficient compressors and turbines with minimum PRc and maximum PRt establishes the minimum, but still finite, MR. Maximum PRc and minimum PRt establishes the maximum MR.

ATR Turbomachinery Design Considerations vs Cruise Mach

Preferred turbomachinery characteristics can vary widely with cruise Mach.

For the low speed mission, Fs can be lower since any inlet spillage drag, which is driven by dynamic pressure, is less at lower speeds. Since Fs may be traded for Isp, a higher Isp is possible. The goal is to maximize mission average Isp without violating the minimum allowable Fs anywhere during the flight. There are several approaches to accomplish this goal that relate directly to the turbomachinery.

For fuel lean operation, a lower PRc will promote a higher Isp and lower Fs. This also increases the size of the combustor because of the reduced combustor pressure and increased total combustor flow rate. This effect can be seen in the following plot of combustor throat area (combustor size indicator) to thrust ratio as a function of PRc. Although this plot is for the high HV propellant, the low HV plot is very similar.

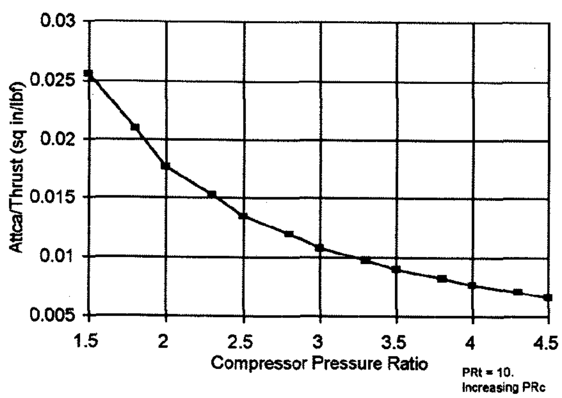

Figure 3. Attca/Thrust vs Compressor Pressure Ratio (High HV)

The lower limit on PRc is established when either 1) the combustor volume becomes excessive or 2) the combustor pressure is so low that it begins to degrade Isp. Vehicle envelope constraints may require an even smaller combustor size which will further increase the minimum acceptable Fs. In all cases, η_c should be as high as possible since reduced η_c reduces Fs **and** Isp.

In the case of the turbine, PRt should be as high as possible, since this promotes high Isp at the cost of reduced Fs. In general, the limiting factor on maximum PRt is the required turbine inlet pressure which is determined by the available GG pressure. In the case of a SFGG, this maximum pressure will be somewhere between 2000 and 3000 psia. Beyond this pressure range, the GG becomes heavy enough to degrade vehicle performance. The effect of PRt on combustor volume is not as great as that of PRc. This somewhat conservative conclusion assumes no "jet pumping" action of the fuel flow on the combustor pressure; Pc is 1) assumed to be driven by the compressor exit pressure and 2) the turbine exhaust pressure can expand as needed to accommodate this pressure.

In the case of the high speed mission, engine size, because of potential spillage, cowl and nacelle drag considerations, becomes crucial. Both the frontal area and combustor size must be reduced to a minimum and yet still be able to produce the required thrust with a reasonably high Isp. In this case, an increased PRc is nearly always required. The turbine can also be used to reduce frontal area. Reducing PRt will increase MR which will reduce the required air flow for a given thrust and thus reduce frontal area. An approach not recommended is to purposefully degrade η_t since thisl increases Fs but also reduces Isp and increases the required turbine inlet pressure. Reducing PRT on the other hand has the same effect but reduces the required turbine inlet pressure.

Any mission requiring a significant acceleration leg further complicates these basic turbomachinery considerations.

Heating Value

Fuel HV is an especially useful parameter for comparing TJ fuels. In the case of the TJ, the combustion gas characteristics like MW and γ vary very little since the combustor gas is typically mostly air. In fact, the combustor MR is almost always operated fuel lean to limit the turbine inlet temperature. In this case, more thrust is available by afterburning downstream of the turbine. However, afterburning also degrades the Isp of the engine. Thus the usual TJ propellant objective is to have a high HV so that the minimum amount of fuel is required to raise the combustor gas temperature which is the major single driver of TJ thrust.

The role of HV in the assessment of ATR propellants is not the same as in TJs because the ATR turbine drive gas does not come from combustion with air. Thus the gas properties (MW and γ) cannot be taken as constants for the purposes of comparing ATR propellants. An engine level comparison of ATR propellants must be made on the basis of Isp and Fs, not on whether the main combustor is operating either fuel rich or fuel lean.

If a propellant has a high HV, it means that MRs will be low. In effect, less of the high HV fuel is needed to achieve the same combustion gas temperature compared to a low HV propellant. However, high HV propellants used in solid propellant GGs generally generate a low MW effluent gas. The lower the MW, the more efficiently the gas will drive a turbine. As the efficiency of the turbine drive gas increases, the less the required fuel flow rate. Thus MR will be low as well. Conversely, for a low HV propellant, MRs will generally be high. However, low HV propellants usually generate a higher MW effluent gas. The higher the MW, the less efficiently the gas will drive a turbine. As the efficiency of the turbine drive gas drops, the required fuel flow rate increases. Thus MR will be high. So, for either high or low HV propellants, MRs and MR are both low or high together. Thus the relationship between MRs and MR can not be reliably predicted on the basis of HV. Therefore, the determination of fuel rich or fuel lean ATR operation is also not possible from a consideration of HV alone.

Two other considerations are noted here. First, the presence of hydrogen in the GG exhaust is often characteristic of high HV propellants. Gaseous hydrogen is a very efficient turbine drive gas, largely due to its low MW. The presence of hydrogen is one reason that MR and MRs are generally low for high HV propellants. Secondly, for some fuels, the turbine drive gas may contain solid carbon (soot). The carbon will increase the predicted HV of the propellant, but it does not expand in the turbine flow passages and thus provides little or no turbine work. This will increase the fuel flow needed to drive the compressor and thus increase MR above the value that would be obtained with complete extraction of work from the drive gas.

From an engine only viewpoint, a fuel rich ATR with a superior Isp and Fs is preferred over a stoichiometric ATR design with inferior Isp and Fs. This author recommends that ATR fuels be evaluated and compared on the basis of their predicted Isp and Fs potential instead of their HV.

SFGG Throttling

There is a definite lower limit that exists for throttling of the SFGG ATR. This limit is reached if the GG valve opens enough so that flow through the valve becomes unchoked. At this point, the turbine throats (fixed) are usually choked and they now determine the GG flow rate. Opening the GG valve beyond this point will have a minimal affect on the operation of the ATR. This lower throttling limit can be increased by designing into the ATR a large total pressure drop between the GG and turbine inlet. This is done by reducing the size of the GG valve throat area relative to the turbine throat area. The greater the desired pressure drop, the greater this relative size difference will be. This assumes that the major portion of the total pressure drop is due to an equivalent normal shock between the GG valve and turbine inlet. Depending on the length/diameter of the duct, there will likely exist a shock train instead a single normal shock. This effective normal shock drops in strength as the GG valve opens because of the reduced expansion seen by the gas as it flows through the GG valve. Generally, total pressure losses are to be avoided. But if significant throttling is required, this total pressure drop must be available to enable the GG valve to control throttling. In this way the valve can be opened significantly (i.e. throttled down) from the design point before the lower throttle limit is reached. The maximum throttling limit is determined by either GG pressure or the physical size of the minimum GG valve throat area. That is, as the GG valve is closed down, GG pressure rises. The maximum GG pressure must not exceed the structural limit of the GG case. The GG pressure rise will be more severe for solid propellants which have a higher burn rate exponent. Therefore, if significant throttling is required, a solid grain formulation with a lower burn rate exponent is preferred to minimize GG pressure variations.

Computer Codes

This section will describe briefly the various computer codes used in the the analysis described in this paper.

ATR Codes. In general there have been 2 approaches implemented in ATR simulation codes.

The first is to specify a MR as well as a compressor design and speed. This means that the code must configure a turbine that will drive the compressor **and** provide the specified MR. This turbine configuration includes number of stages, maximum tip diameter and throat area as well as required inlet pressure. This approach can easily generate unfeasible turbine designs. For example, if a MR of 0.05 is specified and the selected turbine drive gas has a low temperature and high MW, then the resulting turbine will likely have an enormous number of stages in order to extract the needed work from the GG drive gas. Essentially, this approach requires a realistic but very robust turbine design algorithm. Such algorithms are not easy to develop. Despite this drawback, this approach can be useful for some applications.

A second approach, and the one used in the codes developed for this research, assume that both the turbine and compressor are fully characterized, either on the basis of analysis or test data. This approach prevents the designer from selecting the MR but it also prevents the generation of unfeasible turbine designs. It is most appropriate when a selection of compressors and turbines has already been identified and an evaluation/comparison of the different compressor/turbine combinations is to be accomplished. The ATR design code creates a data file that characterizes the ATR design. The ATR off-design code uses this data file to predict off-design operation and performance of the ATR. This off-design data is also stored in a file usable to the trajectory code for simulating operation of the ATR in a flight vehicle. The use of data files, for both overall engine operation as well as component modeling, is considered an important improvement over previous versions in which different subroutines had to be written for each new component. Currently there are data file models for the compressor, turbine, combustor and inlet as well as propellant/combustion related data.

Aerodynamic Coefficient Code. This code is used to predict important aerodynamic coefficients for aerospace vehicles. For this research, the important parameters are lift and drag coefficients as a function of Mach number and angle of attack. The code has been used frequently to accurately predict these, and related, aerodynamic coefficients for a wide variety of aerospace vehicles. In this case, it was use to generate a set of aerodynamic coefficients for a somewhat generic vehicle. The major emphasis was to consistently use this data set for the vehicle flight simulations, regardless of the propulsion system being used. This approach promotes an unbiased comparison of propulsion systems on the basis of vehicle performance, which is the goal of the study.

Trajectory Simulation Code. This code is a FORTRAN code originally developed for use on mainframe computers and later modified for use on a PC. In it's original form it had options for SRM booster engines and other airbreathing propulsion options. It was modified to accommodate ATR and TJ propulsion options. The code is basically batch run and produces output files that can be used to plot both vehicle) and engine parameters as they vary during the trajectory.

Procedure

The general procedure steps implemented in this research are outlined here.

Aerodynamic Coefficients. The first step was to determine the vehicle aerodynamics using the aerodynamic coefficient code. The resulting Cl and Cd values were then put into a data file usable by the trajectory analysis code. This step may be skipped if the needed aerodynamic coefficients are already available. It should be noted here that these coefficients have a tremendous impact on vehicle range and flight time.

Baseline Vehicle Flight Simulation. The second step was to roughly duplicate flight performance of the baseline vehicle. The final results of the propulsion system comparison are always more credible if they have been used to accurately predict the flight performance of an existing system.

ATR Powered Vehicle Flight Simulation. The third step was define an ATR design and analytically fly it in the baseline vehicle.

TJ Powered Vehicle Flight Simulation. The fourth step was to repeat step 3 above using a TJ to power the baseline vehicle.

Propulsion System Comparison. The last step was to compare the relative performance of the three propulsion systems on the basis of range and time of flight.

Analysis

Aerodynamic Coefficients. The vehicle selected is currently powered by a 2 pulse SRM. It has an outside diameter of about 10 inches and weighs approximately 800 lbm at launch when powered by an SRM. It is illustrated in Figure 2 below.

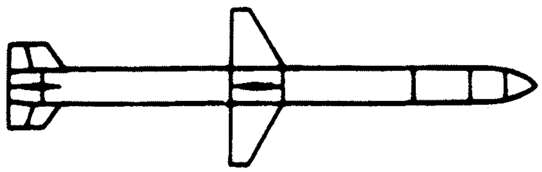

Figure 4. SRM Powered Vehicle

Approximate measurements of the fins was done to provide the needed input data from the aerodynamic coefficient code. The results of the analysis are summarized in the following two figures.

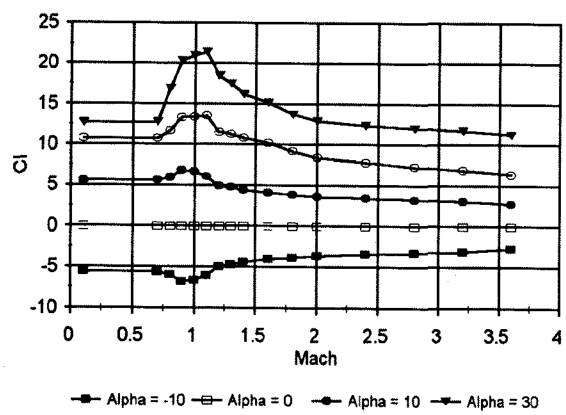

Figure 5. Cl vs Mach and Alpha (baseline vehicle)

Baseline Vehicle Flight Simulation. The baseline vehicle was flown using a two pulse SRM. An estimated weight breakdown for this vehicle is shown in Table I below.

Table I SRM Powered Vehicle Mass Breakdown

Item	Mass (lbm)
Inert	407
Propulsion System	393
Propellant	340
Glow	800
Burnout	460

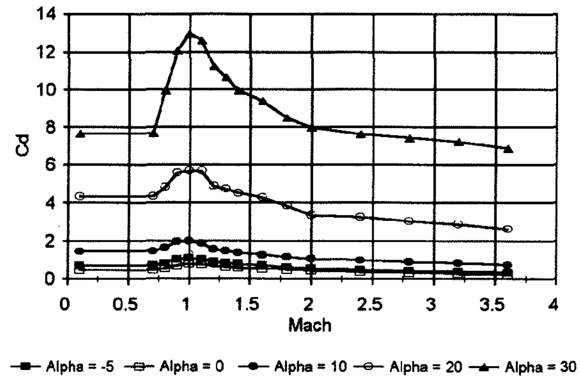

Figure 6. Cd vs Mach and Alpha (baseline vehicle)

The basic mission/flight profile was to first simulate a 3 second ballistic phase. This corresponds to a 3 second drop of the vehicle from the initial flight condition at Mach 0.9 and 10000 feet altitude. The boost thrust was assumed to be about 2500 lbf with a sea level equivalent Isp of 250 seconds. Sustain thrust level was assumed to be 1000 lbf with a sea level equivalent Isp of 235 seconds. For the given propellant mass, the total burn time of the SRM was about 49 seconds. At SRM burnout, the vehicle had reached the cruise altitude of about 12000 feet at a Mach number of 1.82 at a range of about 20 miles. The balance of the flight is basically a dive maneuver. The final range was 28.2 miles with a total flight time of 98.1 seconds.

The following 3 plots show Mach, flow rate and range as a function of time during the flight.

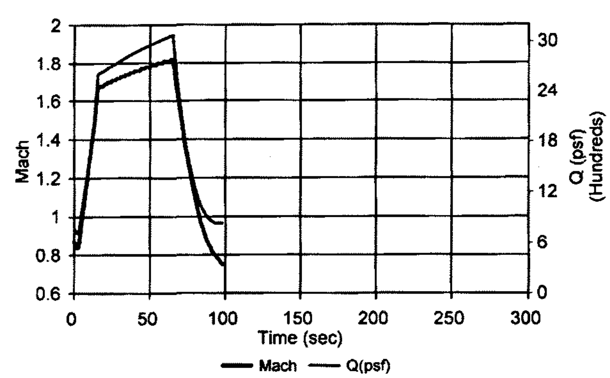

Figure 7. Mach vs Time (SRM Vehicle)

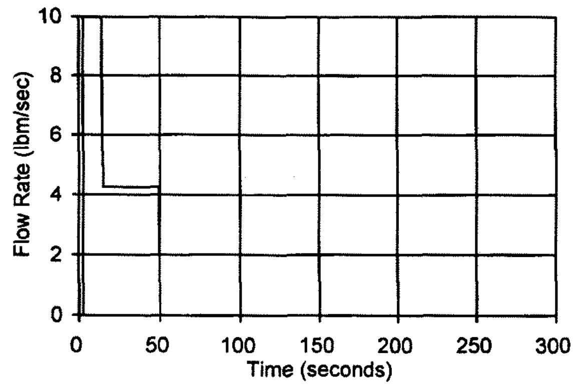

Figure 8. Flow Rate vs Time (SRM)

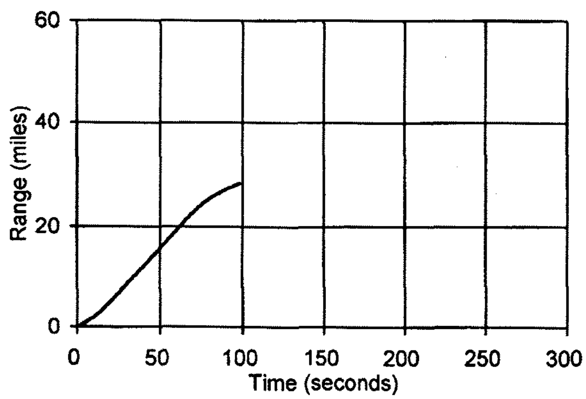

Figure 9. Range vs Time (SRM)

ATR Powered Vehicle Flight Simulation.

Some of the important considerations impacting design of the ATR are noted here.

First, the total volume available for the combination of ATR and propellant is known from the baseline vehicle. This must be allocated between the ATR itself and the SFGG. The ATR propulsion system mass and propellant mass will vary compared to the SRM. This variation is due to 1) engine mass differences, 2) engine and propellant volume differences and 3) propellant density differences. Generally, the combined weight of the ATR plus propellant is less than the SRM because the entire propulsion system volume does not contain propellant.

Also, the vehicle usually has a diameter constraint that will limit how large the burn surface area can be in an end burning SFGG. This effectively defines the maximum turbine drive gas flow rate possible with a reasonable GG pressure (500 to 2800 psia).

Some ATR designs may be eliminated because they simply will not operate over the entire anticipated trajectory.

Fuel selection is another consideration. Some fuels can be eliminated for various reasons, such as:

I) looks feasible analytically but not adequately developed
ii) excessive soot content in exhaust
iii) metal content in exhaust; this can severely degrade turbine performance
iv) burns too hot for use with a turbine
v) MW of exhaust gas is too high (regardless of how low the HV may be, MR will also be very high and Isp will likely be very low)

For the purposes of this study, the high HV propellant noted earlier was selected mostly on the basis of familiarity with it's potential as an ATR fuel.

Compressor selection was limited to a monorotor configuration in which the compressor and turbine are mounted back to back on the shaft. The turbomachinery performance maps assumed for the ATR were also specifically developed for the ATR. The wheel diameter (the same for both the compressor and turbine) is about 7.5 inches.

The weight breakdown for the ATR powered vehicle is shown below.

Table II ATR Powered Vehicle Mass Breakdown

Item	Mass (lbm)
Inert	407
Propulsion System	93
ATR	43
GG	50
Propellant	250
Glow	750
Burnout	500

Several ATR operating parameters are important during engine operation. These include compressor stall margin, GG pressure, fuel and air flow rates, Isp and thrust. All of these parameters, as they varied during the flight, are illustrated in the following plots.

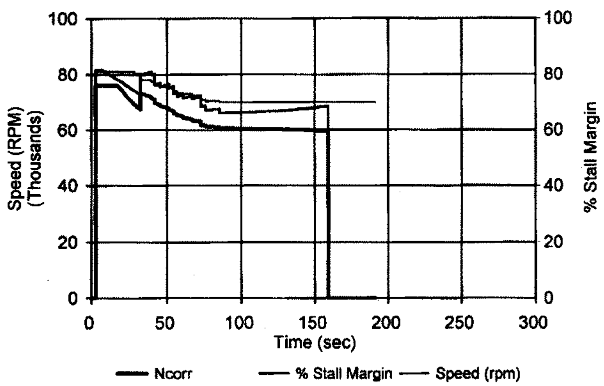

Figure 10. Stall margin vs Time (ATR)

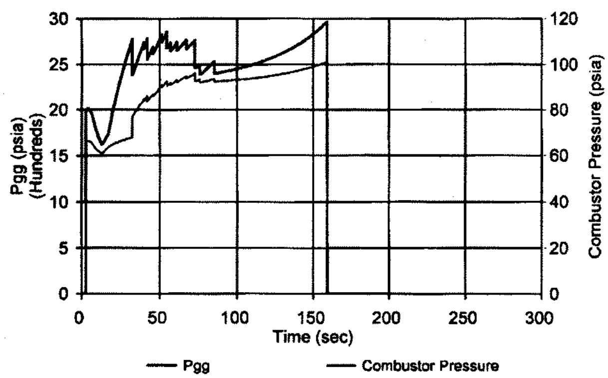

Figure 11. Gas Generator Pressure vs Time (ATR)

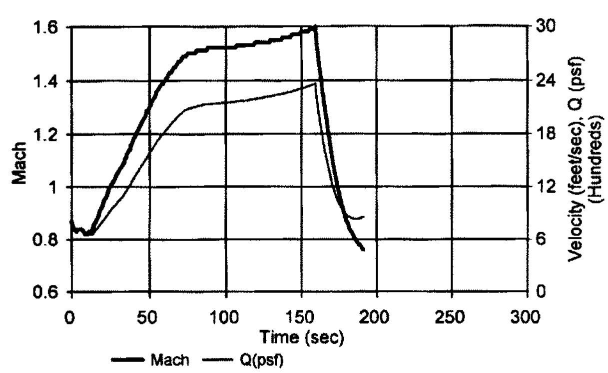

Figure 12. Mach vs Time (ATR)

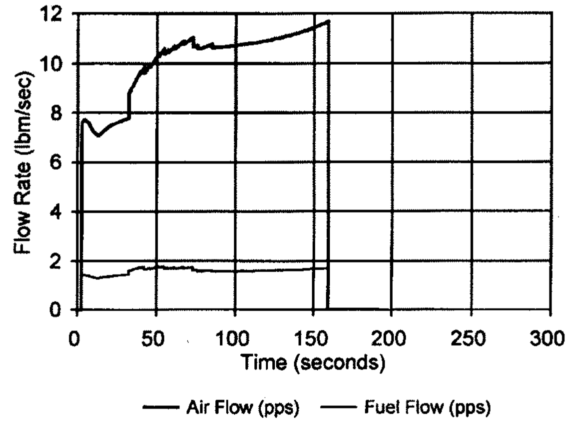

Figure 13. Flow Rate vs Time (ATR)

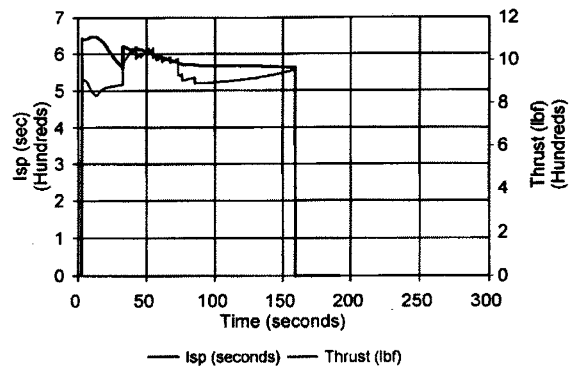

Figure 14. Isp & Thrust vs Time (ATR)

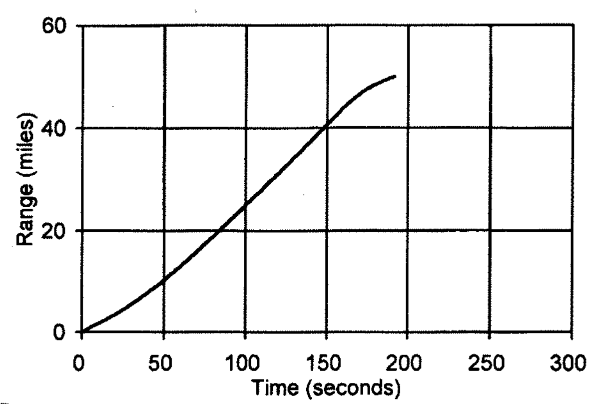

Figure 15. Range vs Time (ATR)

The last plot shows range vs time for the ATR powered vehicle. It shows that the ATR requires a time of flight of about 192 seconds to achieve a range of nearly 50 miles.

TJ Powered Vehicle Flight Simulation.

The TJ selected is actually a scaled up It was scaled by a factor of about 12.5 such that the inlet area would match that of the ATR inlet (20 sq inches). The resulting air flow rates and pressure recoveries, for the flight conditions experienced in the simulations, were similar.

The weight breakdown for the TJ powered vehicle is shown below.

Table III. TJ Powered Vehicle Mass Breakdown

Item	Mass (lbm)
Inert	407
Propulsion System (TJ)	54
Propellant	64
Glow	525
Burnout	461

Note that the vehicle GLOW is greatly reduced compared to either the SRM or ATR powered vehicles. This weight breakdown was arrived at after initially flying the TJ powered vehicle with the available propellant volume completely loaded with JP-10 fuel. The results of this simulation showed that: 1) the vehicle was unable to achieve supersonic flight and 2) operating subsonically, the TJ has, as expected , superior range (over 100 miles) but excessive flight time (nearly 10 minutes). In an effort to make the comparison more unbiased, the TJ propellant was off loaded, as well as some of the inert weight for tankage, leaving enough propellant to achieve the same range as the ATR powered vehicle. The result is the much reduced GLOW shown in the weight breakdown.

The final results for the TJ powered vehicle are summarized by the following figures.

The first figure shows Mach as a function

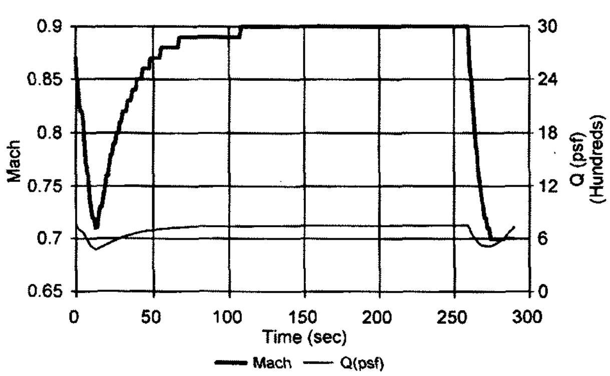

Figure 16. Mach vs Time (TJ)

of time.

The next plot shows air flow as a function of time. This plot can be compared to Figure 13

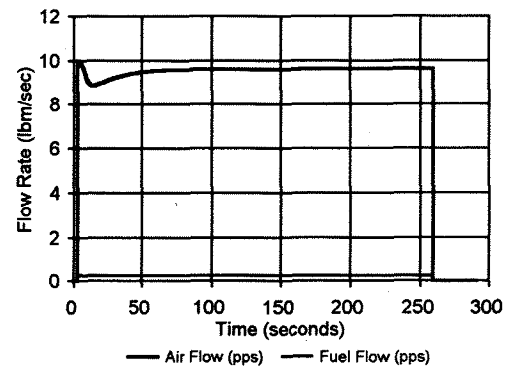

Figure 17. Flow Rate vs Time (TJ)

which shows the ATR air flow rate.

A similar plot for Isp and thrust for the TJ is shown in Figure 18 below. This plot shows that it took the TJ powered vehicle about 100

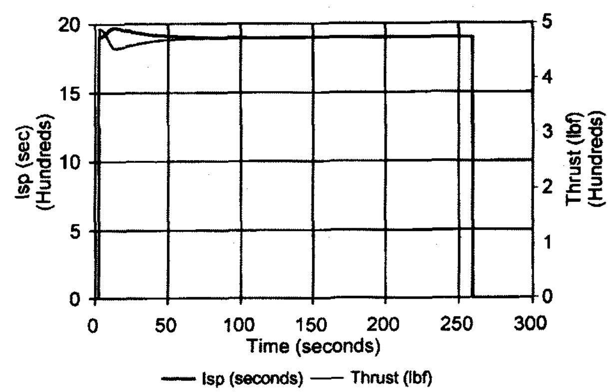

Figure 18. Isp & Thrust vs Time (TJ)

seconds to reach it's maximum Mach of 0.9 at the cruise altitude of 12000 feet.

An important observation is made here regarding the use of this scaled TJ engine. Although it is assumed capable of supersonic flight, it does not represent the latest development in supersonic expendable TJs. Of particular interest is the Fs capability of advanced supersonic TJ designs. For vehicles needing to make the transonic transition, a high Fs will be required. However, as with the ATR, if this improved Fs comes at the cost of an inadequate mission average Isp, vehicle performance improvement may not result.

The final plot shows range of the TJ

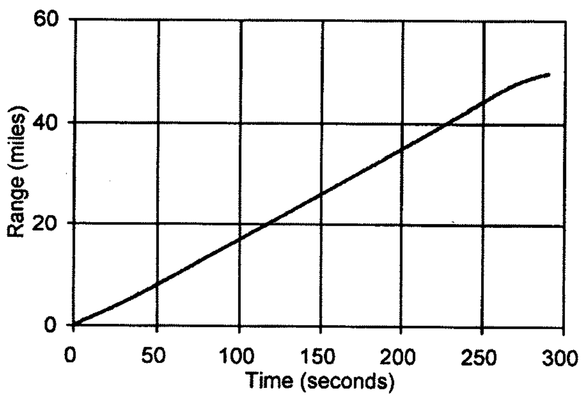

Figure 19. Range vs Time (TJ)

powered vehicle as a function of time. Flying at a maximum Mach of 0.9, the TJ powered vehicle was able to achieve a range of almost 50 miles with a flight time of 290 seconds.

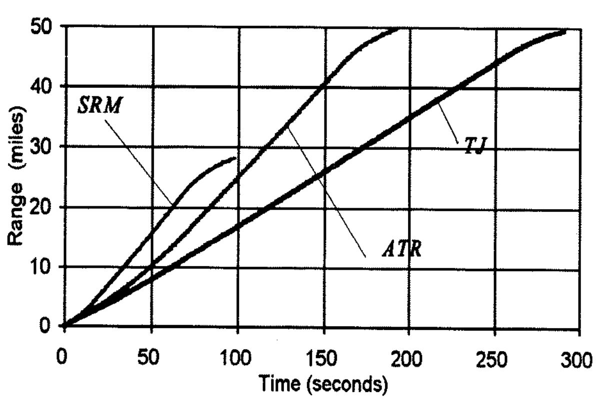

Figure 20. Range vs Time of Flight

Propulsion System Comparison.

A plot, showing the range and flight time, for all three propulsion systems is shown below.

Results/Conclusions

The important results of this study are summarized in the following table.

System	Range (miles)	Time of Flight (sec)	Max Mach	GLOW (lbm)
SRM	28.2	98.1	1.82	807.
ATR	49.8	191.5	1.60	750.
TJ	49.6	290.0	0.90	525.

As this table shows, for the same size vehicle the ATR will provide nearly double the range of an SRM powered vehicle. The flight time is also roughly double that of the SRM powered vehicle. This means that the mission average velocity of the ATR is, as expected, somewhat less than that of the SRM powered vehicle. Compared to the TJ powered vehicle, the ATR achieves the same range in about two thirds of the time compared to the TJ. The weight difference is the penalty paid for the ATR's shorter time of flight.

Conclusions/Recommendations

These preliminary results indicate that the SFGG ATR can provide a reduction in flight time of about one third of the TJ flight time for the same range. This reduced time of flight is at the expense of a 43% increase in vehicle weight, at least for the vehicle evaluated here.

These conclusions could change significantly if the initial Mach was very low. Because of it's higher Fs, the ATR would achieve supersonic flight sooner than the TJ powered vehicle. The reduction in time of flight will likely be more than 33%. A comparison of the TJ and ATR will depend to a large extent on the Isp and Fs capability of the latest supersonic expendable TJ engines. In particular, the dependency of Isp on Mach, which is very pronounced as Mach increases, will have a great effect on both the range and flight time of the supersonic TJ powered vehicle.

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