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EVALUATION OF ON-BOARD HYDROGEN STORAGE METHODS FOR HYPERSONIC VEHICLES

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ABSTRACT

Hydrogen is the foremost candidate as a fuel for use in high speed transport. Since any aircraft moving at hypersonic speeds must have a very slender body, means of decreasing the storage volume requirements below that for liquid hydrogen are needed. The total performance of the hypersonic plane needs to be considered for the evaluation of candidate fuel and storage systems.

To accomplish this, a simple model for the performance of a hypersonic plane follows. To allow for the use of different engines and fuels during different phases of flight, the total trajectory is divided into three phases: subsonic-supersonic, hypersonic and rocket propulsion phase. The fuel fraction for the first phase is found by a simple energy balance using an average thrust to drag ratio for this phase. The hypersonic flight phase is investigated in more detail by taking small altitude increments and calculating the thrust, drag, fuel fraction and the effective specific impulse of each increment. This approach allowed the use of flight profiles other than the constant dynamic pressure flight. The effect of fuel volume on drag, structural mass and tankage mass was introduced through simplified equations involving the characteristic dimension of the plane. The propellant requirement for the last phase is found by employing the basic rocket equations.

The candidate fuel systems such as the cryogenic fuel combinations and solid and liquid endothermic hydrogen generators are first screened thermodynamically with respect to their energy densities and cooling capacities and then evaluated using the above model.

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Introduction

Hydrogen is the foremost candidate as a fuel for use in high-speed transport. The National Aerospace Plane program has been recently initiated by NASA and the Department of Defense (DOD) for developing hypersonic/trans-atmospheric vehicles for takeoff from conventional airport runways to orbit, or for rapid, long-distance, intercontinental aerospace transportation. For this purpose, air-breathing hydrogen-fueled supersonic combustion ramjet (scramjet) engines are being developed for speeds of Mach 5 to 25.

The main difficulty encountered in the use of hydrogen as a high-speed aircraft fuel is the space requirement for its on-board storage. If hydrogen is stored as a liquid, it requires about four times the volume to produce the same amount of energy as conventional fuels. This is especially important for supersonic and hypersonic aircraft which need to have slender designs.

The main aim of the present study is to identify and evaluate the storage media capable of increasing the hydrogen storage density (mass of hydrogen stored per unit storage volume) to a level higher than that of liquid hydrogen (approximately 70 kg/m^3 of hydrogen).

Evaluation Criteria

During hypersonic flight, besides providing propulsion, the fuel has to contribute to structural and engine cooling. In addition, combustibles other than hydrogen in the storage system may serve as rocket fuel in space flight or may be burnt to provide power for the aircraft subsystems. Therefore, the hydrogen storage density, and the heats of combustion of hydrogen and other combustibles in the storage system are important parameters for the evaluation of possible storage systems.

It should be realized that for any improvement in hydrogen storage density a certain penalty has to be paid in terms of increased gross lift-off mass, decreased specific impulse, or increased cost and complexity of tankage, fuel feed systems and technology development. These effects depend on, among others, the flight trajectory, whether the plane will be designed as a launch vehicle or as a hypersonic transport plane, the structural design of the plane, the types of engines to be used, and the switchover Mach numbers for the engines. Only the first two of these effects are considered in the present evaluation. To account for them, differences in the effective specific impulses and the payload capacities are taken to be the additional evaluation criteria.

Since actual design and flight data for the National Aerospace Plane (NASP) do not exist, a need for a mathematical model for the purpose of evaluating the effective specific impulses and payload capacities, especially in the hypersonic range, was strongly felt.

As a result, in order to provide a tool for the comparison of different fuel storage systems, a very simplified model was written for the SSTO flight of the NASP.

Since the main aim is not to obtain a quantitative description of the performance of a hypersonic vehicle but only to compare the performances of two vehicles with different fuel storage systems, no attempt was made to evaluate various structural and flight parameters. Instead, these were left as adjustable parameters. Initially, only the hypersonic segment of the flight trajectory was considered[1].

Simplifying assumptions such as isentropic compression, constant pressure combustion and isentropic expansion to ambient pressure were used in the evaluation of the engine performance. At this step, engine geometry was assumed not to change with the type of fuel or storage system. Thrust-to-drag ratio for a baseline liquid hydrogen plane was left as an adjustable parameter. The required fuel fraction was calculated using the simplified analysis of Jones and Donaldson[2].

The thrust structure mass, thermal protection mass, fuel tankage mass, engine mass and drag for the vehicle were expressed in terms of a characteristic dimension which was iteratively calculated from the specified mean thrust to drag ratio. Conditions at the combustion chamber exit were obtained by interfacing with the NASA/Lewis Chemical Equilibrium Program CEC. Isentropic frozen expansion to ambient pressure was assumed after complete, constant pressure combustion.

Preliminary Results

Percent changes in the payload capacity and the effective specific impulse with respect to the cryogenic LH2 vehicle were computed to aid in the comparison of different fuel systems. The results indicated that if liquid methane is used as fuel, it would result in a 22% less payload capacity and 10% lower effective specific impulse. But, if a mixture of 83% hydrogen and 17% by mass methane is used, the corresponding reductions in payload capacity and effective specific impulse were 3% and 0.7%, respectively. A sample endothermic fuel, cyclohexane, was also tested. If only the hydrogen extracted from cyclohexane is used as fuel negative payloads are obtained. On the other hand, the effective specific impulse increased by 7% due to decreased drag. This indicated that unless the benzene obtained after the extraction of hydrogen from cyclohexane is used as fuel for a later stage of flight, it will not be possible to use cyclohexane as a hydrogen storage medium.

Modified Model

The preliminary results indicated that in order to be able to evaluate complex fuel systems the model should be modified to enable the use of different fuels for different stages of flight. Furthermore, there is a strong indication that the air-breathing propulsion will not be sustained up to orbital velocities and the last stage of flight will be rocket propelled.

The outcome of the considerations is the second generation model. Its information flow diagram is shown in Appendix A.

In this model the total SSTO flight is considered in three stages: (1) subsonic-supersonic AB propulsion; (2) hypersonic AB propulsion; and (3) rocket propulsion. The hypersonic AB phase is further divided into small segments for which a different dynamic pressure and climb angle can be specified. Thus, for this phase a specified flight profile can be approximated.

For the subsonic-supersonic AB propulsion phase, an average thrust-to-drag ratio is specified and the initial to final mass ratio for this phase is calculated using Jones and Donaldson's analysis.

A more detailed analysis is performed for the hypersonic AB propulsion phase in order to obtain an effective specific impulse profile. At the engine inlet, isentropic compression is assumed. Chemical equilibrium conditions for a complete, constant pressure combustion is obtained by calling relevant subroutines of the NASA/Lewis Chemical Equilibrium program (CEC). Exit velocities are then calculated assuming isentropic, frozen expansion to ambient pressure. Engine inlet size is fixed by specifying an initial thrust-to-drag ratio for hypersonic flight and assumed not to change. The thrust structure mass and the engine mass is taken to be proportional to the gross lift-off mass of the vehicle. Thermal protection mass and the mass of propellant tanks are expressed in terms of a characteristic dimension of the vehicle following Dorrington's approach.[3] Equipment mass is assumed to be independent of flight and vehicle characteristics. Total drag is assumed to depend on the square of the characteristic dimension which is in turn a function of the total propellant volume. The mass ratio for each segment is calculated using a modified version of Jones and Donaldson's analysis[2], and therefore requires information on thrust-to-drag ratio. For this reason, the characteristic length is found by an iterative procedure in which the propellant volume is estimated; characteristic length, drag and total propellant mass is calculated and a new propellant volume is obtained.

During the rocket phase extending up to orbital velocity, it is assumed that the specific impulse is constant at 455 seconds and any variation is accommodated in the velocity losses.

Conclusion

A modified computer model is obtained to evaluate relative performances of hypersonic vehicles with different fuel systems.

The candidate fuel systems such as the cryogenic fuel combinations, gelled fuels, and solid and liquid endothermic hydrogen generators were screened thermodynamically with respect to their energy densities and cooling capacities. Some sample results are presented in Appendix A. These systems will be evaluated using the new computer model with respect to their relative payload capacities and effective specific impulses with the possibility of integration of air-breathing and rocket phase propellants. The critical issue is the availability of data on the NASP and the various propulsion systems, especially on the new integrated engines in various stages of development. For example, cryogenic fuels may start the air-breathing propulsion with LACE engines (or its derivatives) while endothermic hydrogen generators will probably employ engines with some type of turbine propulsion. Also, combustion of certain fuels in some engine types may not be technically feasible. Integration of the cooling duty to propulsion requirements should be the topic of further study.

Bibliography

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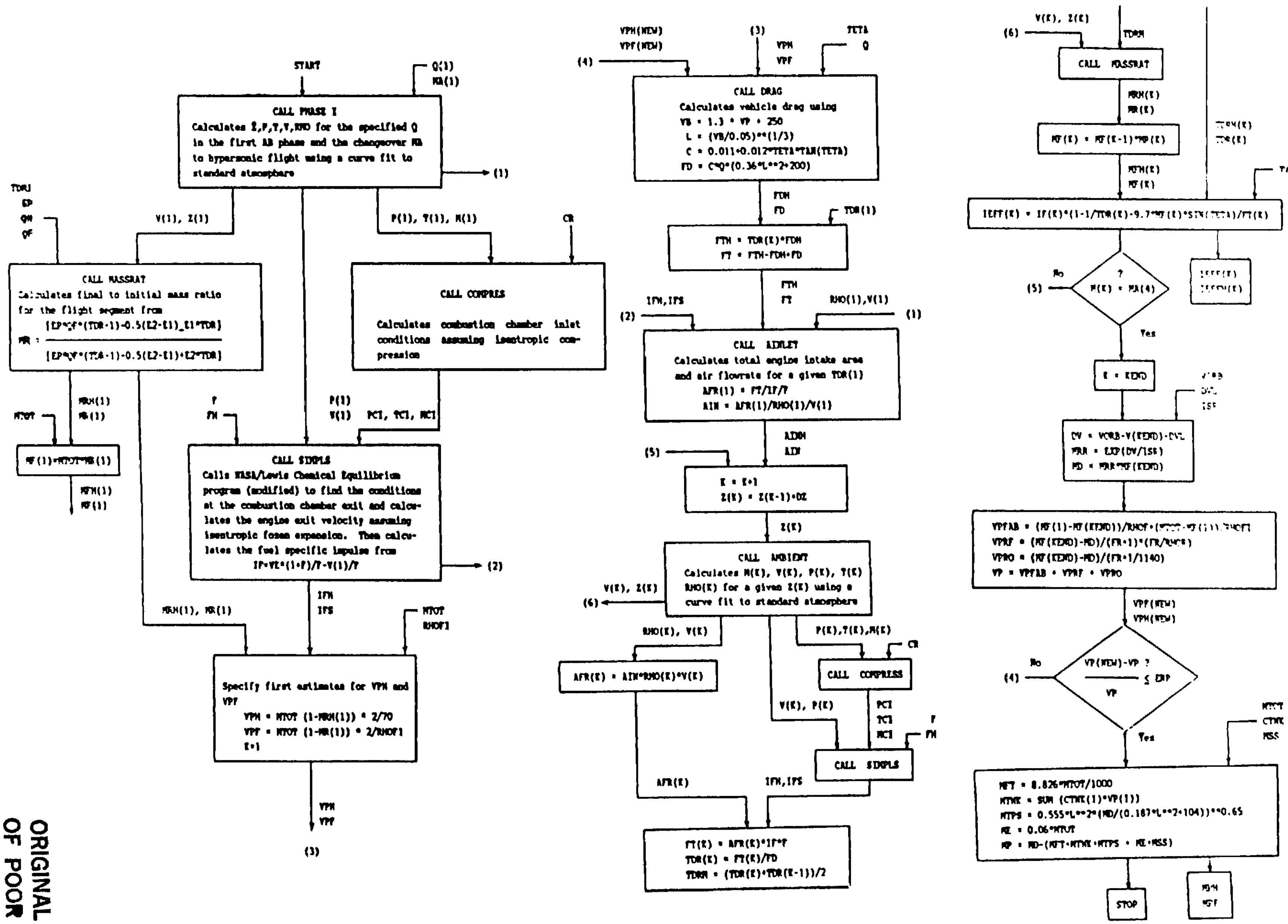
Nomenclature

AFR	Air flow rate for the vehicle with candidate fuel system
AFRH	Air flow rate for LH2 vehicle
AIN	Engine inlet area for the vehicle with candidate fuel system
AINH	Engine inlet area for LH2 vehicle
C	Drag coefficient for the planes (a function of TETA only)
CR	Air inlet compression ratio
DV	Velocity increment for rocket propulsion phase
DZ	Altitude increment
E2,E1	Final and initial total energies for a flight segment defined by E = V2/2 + gZ
EP	Propulsion efficiency
ERP	Error criterion for the convergence of propellant volume calculations
F	Air to fuel mass ratio for the vehicle with the candidate fuel
FD	Drag for the vehicle with the candidate fuel system
FDH	Drag for the LH2 vehicle
FH	Air to fuel mass ratio for the LH2 vehicle
FT	Thrust of the vehicle with the candidate fuel system
FTH	Thrust of the LH2 vehicle
IEFF	Effective specific impulse for the vehicle with the candidate fuel system
IEFFH	Effective specific impulse for the LH2 vehicle
IFH	Fuel specific impulse for the LH2 vehicle
IFS	Fuel specific impulse for the vehicle with the candidate fuel system
ISR	Specific impulse for the rocket propulsion phase
L	Characteristic length of the vehicle
M	Mach number
MA	Switchover Mach numbers
MCI	Mach number at the combustion chamber inlet
MD	Dry mass of the vehicle
ME	Engine mass
MF	Vehicle mass with the candidate fuel system
MFH	LH2 vehicle mass
MFT	Thrust structure mass
MPF	Payload mass of the vehicle with candidate fuel system
MPH	Payload mass of the LH2 vehicle
MR	Final to initial mass ratio of the vehicle with the candidate fuel system, for a specific flight segment

MRH	Final to initial mass ratio of the LH2 vehicle for a specific flight segment
MRR	Final to initial mass ratio for rocket propulsion phase
MSS	Equipment mass
MTNK	Propellant tank mass
MTOT	Gross lift-off mass
MTPS	Thermal protection mass
P	Pressure
PCI	Combustion chamber pressure
Q	Dynamic pressure
QF	Heat of combustion of propellant
RHO	Density
RHOF	Density of the candidate fuel for hypersonic AB propulsion
RHOF1	Density of the fuel for subsonic-supersonic AB propulsion
T	Temperature
TCI	Temperature at combustion chamber inlet
TDR	Thrust to drag ratio
TDRM	Mean thrust to drag ratio
TETA	Climb angle
V	Velocity
VE	Exit velocity
VORB	Orbital velocity
VPF	Propellant volume for the vehicle with the candidate fuel system
VPFAB	Propellant volume for AB propulsion
VPH	Propellant volume for the LH2 vehicle
VPRO	Rocket phase oxidant volume
Z	Altitude



Appendix A

Information Flow Diagram of the Computer Program for the Evaluation of the Performance of Hypersonic Vehicles.