NASA/CR- -92- 207458

7N-20-CR

RI/RD90-149-10

OPERATIONALLY EFFICIENT PROPULSION SYSTEM STUDY(OEPSS) DATA BOOK

Volume X Air Augmented Rocket Afterburning

30 October 1992

Prepared for
Kennedy Space Center
NAS10-11568 (Mod. 8)

Prepared by
Shahram Farhangi

Rocketdyne Study Managers: R. P. Pauckert/G. S. Waldrop
NASA, KSC Study Manager: R. E. Rhodes

Rockwell International
Rocketdyne Division
6633 Canoga Avenue
Canoga Park, CA 91303

FOREWORD

This document is part of the final report for the Operationally Efficient Propulsion System Study (OEPSS) conducted by the Rocketdyne Division of Rockwell International. The study was conducted under NASA contract NAS10-11568, and the NASA Study Manager was Mr. R. E. Rhodes. The Rocketdyne Program Manager was R. P. Pauckert, the Deputy Program Manager was G. Waldrop, and the Project Engineer was T. J. Harmon. The period of study was from April 1989 to October 1992.

ABSTRACT

A study was directed towards assessing viability and effectiveness of an air augmented ejector/rocket. Successful thrust augmentation could potentially reduce a multi-stage vehicle to a single stage-to-orbit vehicle (SSTO) and, thereby, eliminate the associated ground support facility infrastructure and ground processing required by the eliminated stage. The results of this preliminary study indicate that an air augmented ejector/rocket propulsion system is viable. However, uncertainties resulting from simplified approach and assumptions must be resolved by further investigations.

Internal Letter

Rockwell International

Date: 6 March, 1991

No: CDR-91-099

TO:
G. S. Wong
R. Paukert
IB05/IB43

FROM:
S. Farhangi
D/529
IA06
4423-Plummer

Subject: **Rocket Engine Nozzle Afterburning (Ejector/Rocket) Effectiveness for Advanced Launch Systems Applications - (Operationally Efficient Propulsion System Study Task 4.0)**

Reference: (1) Foster, R. W., Escher, W. J. D. and Robinson J., "Air Augmented Rocket Propulsion Concepts," AFAL, TR-88-004, January 1988.

(2) Escher, W. J. D. and Flornes, B. J., et al, "A Study of Composite Propulsion Systems for Advanced Launch Vehicle Applications," Final Report on NASA Contract NAS7-377, The Marquardt Corporation, April 1967.

(3) Gordon, S. and McBride, B. J., "Computer Program for Calculation of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks, and Chapman-Jouguet Detonations," NASA SP-273, March 1976.

(4) Odegaard, E. A. and Stroup, K. E., "Advanced Ramjet Concepts," The Marquardt Corporation Technical Report AFAPL-TR-67-118 Volume VIII, January 1968.

(5) Burkardt, L., Preliminary report of Ejector Computer Program, developed at NASA Lewis Research Center.

(6) Mossman, E. A., Chapman, R. L. and Rozycki, R. C., "Experimental and Theoretical Investigation of the Rocket Engine - Nozzle Ejector (RENE) Propulsion System," AFRPL-TR-65-66, April 1965.

(7) Mehegan, P., "Private Communications," Rocketdyne, October 1990.

1. SUMMARY

This study was directed towards assessment of viability and effectiveness of an ejector/rocket, or rocket engine nozzle after-burning concept. The focus was on performance enhancement and thrust augmentation aspects of an ejector/rocket system for ALS type vehicle and its effect on overall vehicle/propulsion system such as payload weight, Gross Lift-Off weight, and propellant weight.

Ideal flow analyses were conducted and a simple fixed geometry shroud design was optimized to operate as an ejector system in the low speed regime (flight Mach 0 to 2) to augment ALS rocket engine thrust. An ejector with secondary inlet area of 80 ft**2 with area ratio of 1.63 was selected which produced substantial ideal thrust augmentation at all flight Mach number

range of 0 to 2 when utilizing just rocket engine excess fuel (w/o injection of additional fuel). This resulted in maximum payload increase in excess of 27% with ALS fixed vehicle size, or Gross Lift-Off Weight (GLOW) and propellant weight reduction in excess of 19 & 23% respectively with constant ALS baseline payload of about 120 Klbs. and a closely matched flight trajectory for ejector/rocket system. Increase in I_{sp} of about 40 sec would be required to obtain the same (27%) payload increase for ALS.

Effect of combustion of injected fuel, in addition to engine excess fuel, with ingested air was also investigated. The ejector design was slightly modified from the design selected for operation with combustion of rocket engine excess fuel only, in order to prevent thermal choking of ejector flow (inlet area of 80 ft**2 and area ratio of 1.92). Payload increase in excess of 33% with ALS fixed vehicle size, or Gross Lift-Off Weight (GLOW) and propellant weight reduction in excess of 19% & 22% respectively with constant ALS baseline payload were achieved. Ideally the effect of injection of additional fuel will be more pronounced with a more closely matched flight trajectory and even more with an ejector geometry more suitable for fuel addition flying an appropriate flight trajectory.
Based on the result of this preliminary study it is concluded that an ejector/rocket propulsion system is a "viable" system and can be effectively utilized for ALS type missions. However, uncertainties, resulting from simplified approach and assumptions in regards to effectiveness of an ejector/rocket system must be resolved by further investigations. Major issues and required efforts are identified and a more comprehensive study is essential and highly recommended.

2. INTRODUCTION/BACKGROUND

The simplest form of air-augmentation of a rocket propulsion system (Ref. 1 & 2) is to install a simple geometry light weight extension of the rocket engine nozzle. The air augmented shrouded rocket concept is basically a conventional rocket engine (like ALS) shrouded by a simple ejector which captures, directs and mixes atmospheric air with the rocket nozzle exhaust gas. The air augmentation is the ingestion, compression and mixing and combustion of air with exhaust gas (and/or additional fuel); so that the specific impulse of the system increases. This concept is promising since the all rocket propulsion systems have excess fuel in their exhaust gas and if the otherwise wasted chemical energy of this fuel contained in the mixture of fuel rich exhaust (or added fuel) is combusted with the ingested atmospheric air and further expanded in a divergent section , additional thrust and increased I_{sp} is produced from additional expansion surfaces.

Figure 1 illustrates ejector/rocket propulsion system concept. The conventional bell-nozzle of a rocket engine (ALS) is surrounded by an ejector consisting of the air inlet and a divergent mixing/after-burning chamber. The two streams, primary formed by rocket exhaust and secondary stream consisting of atmospheric air begin to mix at the exit of rocket engine nozzle and combust (with or without additional injected fuel), with relatively low secondary to primary mass flow ratio supersonic exhaust appears to exist. In the mixing process, part of the primary stream's high kinetic and thermal energy is transferred to the secondary stream by direct momentum exchange. Additional thermo-chemical energy is released by combustion of fuel rich exhaust and/or injected fuel. In the process of energy exchange, the momentum flux of the fluid increases and produces useful thrust.

In order to achieve efficient mixing and reaction, flow in combustors for usual ducted rockets, ram-rockets, and ejector jet systems must be subsonic. Re-acceleration of the exhaust to supersonic speeds requires second throat and considerable additional volume. However, if mixing and reaction could be accomplished in supersonic nozzle flow, the augmentation can be realized in spite of the lower cycle efficiencies (high entropy rise/high total pressure loss) associated with supersonic mixing and after-burning.

3. OBJECTIVE

The purpose and scope of this study is to perform simplified analysis to determine viability of Rocket Engine Nozzle After-burning concept and to identify an ejector geometry envelop suitable for operation in the range of flight Mach numbers of 0 to 2. Inviscid flow and ideal pumping, mixing, combustion, with no aerodynamic losses is assumed to conduct the analyses. In order to eliminated geometric (design) complexity and increased volume/weight associated with second throat and expansion surface, the secondary air flow is limited to a level that supersonic mixing and combustion is achieved with a diverging ejector geometry. The ejector is designed as a simple fixed geometry shroud and therefore its inlet kinetic energy efficiency (total pressure losses) limits its effectiveness above Mach 2. The inlet of this simple fixed geometry ejector, assuming ideal ejector pumping, will limit secondary air flow rate to achieve supersonic mixing and combustion in the ejector.

4. OVERALL APPROACH

ALS type vehicle and flight trajectory is used to determine an ejector geometry that provides significant thrust increase for a single Space Transportation Main Engine (STME). In order to define an ejector geometry envelop suitable for operation in the range of flight Mach number of 0 to 2, an attempt is made to identify the optimum ejector geometries (point designs) for static operation, flight Mach numbers of ~0.45, ~0.8, ~1.0, and ~2 conditions. These optimum geometries provide maximum ejector/rocket thrust at their respective flight speeds based on ALS all rocket flight trajectory without any additional injected fuel. Weights of these optimum geometries then are estimated based on existing available ALS nozzle weight data. Mission analysis is performed with ALS reference (baseline) vehicle and payload capacity is determined with and without ejector installed on all engines. The increased thrust and specific impulse with ejector is traded off against the resulting increase in drag and weight. The ejector geometry that results in maximum payload increase is then selected as the ejector baseline geometry. Performance or thrust of ejector/rocket propulsion system, unlike all rocket system, depends not only on altitude but on flight speed also, therefore initial flight trajectory requires some modifications. A new flight trajectory then is identified that more closely matches altitude/thrust/flight Mach number and new calculated thrust is used to perform mission analysis again. This iteration process will eventually (in 2-3 iterations) converges and results in matched altitude/thrust/flight Mach number. The effectiveness of ejector is then determined based on payload increase with fixed vehicle size or GLOW & propellant weight reduction with fixed payload weight.

This ejector geometry is also used to determine the effect of injection of additional fuel on ejector performance. Slight modification to ejector base-line geometry (area ratio) is required to eliminate possibility of thermal choking of ejector flow, if fuel is to be added continuously from static to Mach 2 operation. Again the payload increase or GLOW & propellent weight reduction is determined by mission analysis. The iteration process to obtain a solution that is achieved by matching trajectory with performance/thrust level is again necessary.

5. TECHNICAL APPROACH

Ejector performance/thrust is affected by ambient air (free-stream) condition, flight velocity, secondary flow condition (inlet geometry and pumping capability), primary rocket flow thermo-chemical condition, energy released by combustion of ingested air with fuel (nozzle exhaust excess w or w/o additional injected) and ejector geometry. Most importantly, ejector performance depends on the level of mixing between primary and secondary flow and combustion of secondary air with additional injected and/or rocket exhaust excess fuel. The pumping capability of an ejector depends on the level of mixing between the two streams and therefore mixing and pumping are inter-related and, especially at low speeds, any change in the mixing level directly affects pumping and vice versa.

This preliminary study was conducted with certain assumptions to simplify calculations. The simplified approach was taken to eliminate tedious and time consuming, sophisticated/advanced calculation techniques, yet perform first level analysis to assess viability of an ejector/rocket propulsion system. One dimensional inviscid, ideal flow with equilibrium chemistry and jumped (path independent) calculation was conducted. Therefore all the effects of flow multi-dimensionality, non-uniformities (pressure, temperature, velocity, Mach number and chemical composition), viscosity, incomplete mixing & pumping, and chemical kinetics are neglected. Losses associated with shocks due to flow interactions, velocity vector (divergent), incomplete mixing and combustion, wall heat transfer and internal drag are not accounted for, and it is assumed that mixing and combustion is completed (equilibrium) at ejector exit. For simplicity, the effect of boundary layer (developed on the air induction system wall and on primary nozzle wall) on ejector mixing/pumping and performance along with base flows and nozzle lip effects are neglected. Flow separation in the primary nozzle and ejector section due to adverse pressure gradient and shock boundary layer interaction is also neglected and the system is flowing full.

5.1 Ejector Thrust Calculation

In order to determine ideal thrust generated by ejector/rocket, ambient, primary and secondary flow conditions at the plane where mixing starts (station 1 Figure 1), and ejector geometry must be known. Primary flow condition, at the rocket engine nozzle exit was determined based on STME GG cycle Main Combustion Chamber (MCC) data. Engine data, chamber total pressure of 2250 psia and mixture ratio (MR) of 6 with fuel (H2) temperature of 190 ^{o}R and LOX temperature of 170 ^{o}R, were used and the flow was expanded with equilibrium chemistry to nozzle area ratio of 40 (e=40) to determine rocket engine nozzle exit flow conditions (Ref. 3). The secondary air flow conditions were determined based on free-stream static pressure, temperature and flight velocity and in subsonic flight regime the secondary inlet flow was assumed choked (M_S=0.9) and at supersonic flight speed(s) (M_O=2) it is shocked down to subsonic flow. Isentropic inlet process determines secondary flow conditions at subsonic flight speeds, but in order to account for inlet total pressure loss (entropy rise) at flight Mach 2 the free-stream total pressure was adjusted according to inlet kinetic energy efficiency reported by Marquardt on ejector/ramjet test (Ref. 4).

Mathematically, the ejector is described by applying the various Conservation Laws, along with the Equation of State for ideal gases, between the two defined stations 1 & 2 (beginning of mixing and ejector exit) as shown in Figure 1. One dimensional equilibrium ejector code developed by Dr. L. Burkardt at NASA LeRc (Ref. 5) was modified and used to facilitate ejector thrust calculations. Ejector wall pressure force is determined by linear pressure distribution along the flow axis assuming the inlet wall pressure is due mostly to secondary stream.

The calculated ejector/rocket thrust includes air inlet ram drag and total pressure losses at Mach 2. The resultant ejector thrust is normalized by the rocket thrust operating at the same ambient condition and thrust augmentation (Aug) represents increase in the rocket thrust.

5.2 Mission Analysis

Ejector performance benefits were estimated by running trajectories for typical ALS vehicles and comparing changes in payload capability and/or gross weight. Two vehicles were considered, one having a reference payload of about 146,000 lbs. and a more recent design with about 120,000 lbs. payload. As ground rules ALS vehicles were two-stage parallel burn with seven Liquid Rocket Booster (LRB) engines and three core engines. All comparisons are made with one engine out and with ALS nominal orbit of 28.5 degree 80 X 150 n.mi. The thrust increase is modeled as a function of altitude form lift-off to about flight Mach 2 while vehicle reference area increases by ejector inlet area and then ejectors are jettisoned.

Vehicle data for the two ALS configurations were based on data obtained form one of the vehicle contractors for typical designs. Weight breakdowns were available with sufficient detail to allow scaling to new sizes and accounting for the mixture ratio changes in the cases with fuel addition in the ejectors.

Trajectories considered of an eight-second vertical rise, followed by an instantaneous kick-over and a gravity turn which terminated when the dynamic pressure fell to 5 psf. An optimal-pitch profile was then followed to the perigee of 80 X 150 n.mi. The ejectors were used from lift-off to about Mach 2 and then jettisoned.

Ejector performance was modeled as a function of altitude based on a reference (ALS all rocket) trajectory. Since the ejector performance varies with both altitude and Mach number, this simplification introduces an inaccuracy if the Mach number-altitude profile vary significantly from reference trajectory. The engine performance with ejectors, therefore, is re-calculated in an iterative process to match the performance to the trajectory.

5.3 Ejector Weight Estimate

Ejector performance level is a strong function of shroud length. The longer the length of an ejector the more complete mixing of primary and secondary flow, pumping, and combustion, but also increased weight and large ejector volume. The weight, therefore, has to be traded of against increase in performance (thrust). Rocket Engine Nozzle Ejector (RENE) experiment (Ref 6) results indicate that an ejector length equivalent to 1 to 2 times ejector initial diameter is sufficient for application at flight Mach 2. Ejector length equal to ejector initial diameter ($L/D_1=1$) is selected for this study to determine ejector weight.

Ejector weight is estimated based on ALS available rocket nozzle weight data. ALS Gas Generator (GG) cycle nozzle weight breakdown is for average jacket thickness of 0.065 inch. Considering the short duration that ejector is being used and since it is mostly exposed to a cooler gas (ingested air) compared to rocket nozzle, 0.075 inch thick stainless steel was selected (Ref. 7) with ejector length equal to the initial diameter as a basis to calculate ejector weight. A factor of two (2) to ejector shroud weight was applied to account for the attachments and supports weights. Variation of estimated ejector shroud weight, using stainless steel, with secondary inlet area is shown in Figure 2. Advanced exotic material can be considered as an alternative to stainless steel in order to reduce ejector weight significantly.

6. DISCUSSION AND RESULTS

As previously stated the ejector performance is mainly influenced by ambient, primary & secondary flow conditions at the plane where mixing begins and by the geometry of ejector shroud. The primary and secondary flow conditions are determined from Main Combustion Chamber (MCC) and flight free-stream conditions. To select the ejector geometry, parametric studies were performed to determine ejector inlet and exit areas that produced maximum thrust augmentation at a specific point of flight trajectory by utilizing rocket engine excess fuel only (without any additional injected fuel). This parametric study resulted in an ejector design optimized for operation at sea level (M_O=0) with inlet area of 80 ft**2 and ejector area ratio of about 1.63 (the minimum area without thermally choking ejector flow), Figure 3. The ejector geometry optimized for static operation, then was used to determine its thrust augmentation up to flight Mach number of 2. The resultant thrust augmentations (ratio of ejector thrust increase to rocket thrust), achieved without injection of additional fuel, also represent increase in I_{sp}. Thrust augmentation achieved at static condition is in excess of 12% as shown in Figure 4, and despite increase in ram drag (Figure 5), augmentation increases in subsonic regime as flight Mach number is increased, but due to low secondary mass flow (Figure 6) and high induction system (inlet) losses ejector effectiveness at flight Mach 2 is significantly reduced. Inlet designed to operate efficiently at Mach 2 (8% total pressure loss according to MIL-E-5008B) increases ejector thrust augmentation by more than a factor of 2.

A larger ejector, as expected, with higher secondary to primary mass flow rate increased thrust augmentation at higher flight speed while reducing thrust augmentation at static operation. An ejector geometry with inlet area of 125 ft**2 and area ratio of slightly less than 2.0 provided maximum thrust augmentation and Isp increase at flight Mach number of 0.45, Figure 7. The ejector with inlet area of 125 ft**2 and area ratio of 2.07 (area ratio increased from 1.99 to prevent thermal chocking at sea level) produced lower thrust at sea level operation and higher thrust at flight speeds above Mach 0.45 compared to smaller ejector designed for Mach 0 as shown in Figure 8.

Another set of parametric study was conducted to obtain an ejector geometry optimized for flight Mach number of 0.8 to achieve greater thrust augmentation at higher speeds. It is a trade between using a smaller ejector size that will provide high static thrust augmentation and reduced thrust augmentation at high flight Mach numbers, and a larger ejector size that will provide higher thrust augmentation at high flight speeds and reduced static performance. In fact an ejector geometry designed for Mo=0.8 will require an inlet area in excess of 200 ft**2 and will result in excess of 10 % loss in rocket thrust (-10% Aug with 200 ft**2 inlet) at lift off. In addition the ejector becomes extremely large, heavy and impractical. It was then concluded that ejectors with inlet area in the range of 50 ft**2 (slightly smaller than Mo=0 design) to 160 ft**2 (slightly larger than Mo=0.45 design) would be proper candidates for the baseline ejector geometry. Thrust augmentation level of ejectors with inlet area in the range of 80 to 140 ft**2 is shown in Figure 9.

In order to determine an optimum ejector size and geometry, mission analysis was conducted for the range of ejector inlet areas (As=50 to 160 ft**2) and area ratios (ARej=1.15 to 2.26) using baseline ALS vehicle with payload of about 146 Klbs. The increased thrust and specific impulse obtained with the ejector were traded off against the resulting increase in external aerodynamic drag and weight. The results of trajectory analysis are presented in Figure 10. Constant maximum payload gains, in excess of 16% was obtained with fixed vehicle size and ejector geometries with inlet areas of 80 to 120 ft**2. Ejector with inlet area of 80 ft**2 and area ratio of about 1.63, smallest with near maximum payload gain, is selected as baseline geometry. The Gross Lift-Off Weight and propellant weight reduction of 9.6% & 11.8% respectively can be achieved with baseline ejector geometry and fixed ALS baseline payload of about 146 Klbs. An isolated case with the ejector length equal to twice the initial diameter was also examined. The result indicates a reduction of about 2% in payload increase (~14%) compare to the ejector length equal to one initial diameter.

The baseline ejector geometry with inlet area of 80 ft**2 and area ratio of 1.63 was used to determine effect of combustion of additional injected fuel on ejector performance flying reference trajectory. Ejector flow thermally chokes if sufficient amount of fuel is added, in addition to rocket exhaust excess fuel, to burn all the ingested air. In order to prevent thermal choking, the ejector area ratio must be increased and/or the amount of additional fuel must be reduced or fuel must be added at properly selected flight Mach numbers only (above Mach 0.5 with F=1). For an ejector with secondary inlet area of 80 ft**2 (baseline), the ejector area ratio must be increased to 1.92 (from 1.63). Thrust augmentation and increase in Isp of ejector with area ratio of 1.63 with fuel addition at Mach 0.8 and 1 only (scheduled fuel addition) and with area ratio of 1.92 with continuous fuel addition (F=1) are shown in Figures 11 & 12 with their respective gains in Figures 13. The estimated payload gain with fixed vehicle size and reduced GLOW & propellant weight with fixed payload of about 146 Klbs. indicate continuous fuel addition, even thought requires larger and heavier ejector, is more beneficial than scheduled fuel addition.

Ejector effectiveness with the latest ALS baseline vehicle data, obtained from vehicle contractor, was also determined based on reference trajectory (ALS all rocket). This latest baseline vehicle design is a lighter vehicle with baseline payload of about 120 Klbs. Similarly, this vehicle was used for further comparisons and Figure 14 shows the payload improvement as a function of inlet area. Once again the 80 ft**2 inlet area was near optimum and payload increased by about 25%. As the result of more favorable thrust to weight ratio, higher payload gains and reduction in GLOW & propellant weight were achieved with the vehicle design with about 120 Klbs. baseline payload as shown in Figure 15. However, effect of fuel addition was less pronounced for lighter vehicle than heavier vehicle, Figure 16.

Performance of an ejector/rocket system is dependent on the flight trajectory and flight trajectory is dependent on performance level of propulsion system. Therefore, to determine ejector effectiveness accurately, flight trajectory must be matched with thrust level, either by iteration or using maps of performance v.s altitude and Mach number. The lower-payload (120 Klbs.) ALS vehicle was used and engine performance with ejectors was re-calculated in an iterative process to match performance (thrust) to the trajectory (altitude and Mach number). In the cases examined the vehicle with ejectors reached higher Mach numbers at lower altitude (higher dynamic pressure trajectory) compared to the reference flight trajectory due to the increased thrust, Figure 17. This would result in higher ejector performance (Figure 18) so the original predicted payload gains and GLOW reductions were somewhat conservative (Figure 19). One iteration resulted in fairly close match between performance and trajectory, and trajectory nearly converged for cases without fuel addition, Figure 20. This trajectory was also used to determine the effect of ejector length (weight) on ejector/rocket effectiveness. Ejector length was varied and payload increase was estimated for ejector L/D of 2, 3, and 5 and the result is presented in Figure 21. Even though the payload increase was reduced as the ejector length increased (from value of 27.7% for L/D=1 to about 21% for L/D of 5) still significant payload increase was achieved with all the ejector sizes examined.

With first iteration trajectory for the cases with injection of additional fuel, increased thrust augmentations were observed (Figure 22), however due to mismatch between performance and trajectory, the calculated payload increase is somewhat optimistic and GLOW & propellant weight reductions are somewhat conservative (Figure 23). Additional iteration is required in order to obtain a converged solution with matched ejector performance/flight trajectory.

7. KEY ISSUES

7.1 Operation Range/Design

The design complexity of an air augmented rocket system is primarily dependent on the range of flight Mach number that ejector operation and rocket thrust augmentation is desired. It seems logical that the operating range of such a system be established by determining cost of putting a payload into an orbit. Since a variable geometry shroud might be required if rocket thrust augmentation extends over a wide range of operation, net vehicle thrust increase must be traded off against design complexity & system weight.

A simple fixed geometry ejector can operate up to flight Mach number of about 2-3. The shroud then may be jettisoned or, if nozzle pressure ratio is high enough and the level of design complexity is acceptable, can be attached to the nozzle and be used as an extension of rocket engine nozzle. It will be advantageous to use a lower area ratio rocket nozzle (lower weight) if this option is exercised, since usually rocket nozzle exhaust flow is over-expanded at low flight speeds.

The system can also be designed to operate over wider range of flight, from take-off to flight Mach number of about 6 (ejector/rocket - ram/rocket - all rocket). This will require either a variable geometry shroud to allow for efficient air induction and mixing (exchange of momentum) or thrust augmentation level will be low and even undesirable in off design range of flight. Again at flight speeds over Mach 6 for all rocket operation the shroud could be attached and used as extension of rocket nozzle to increase nozzle performance (higher area ratio) during high altitude flight, or jettisoned.

For existing all rocket propulsion systems, shroud design can be tailored to enhance rocket thrust/performance with no or minimal changes to the system hardware. However if the concept is being considered for a new engine, the propulsion system (ejector/rocket or ram/rocket or..) and the vehicle as a whole must be designed to provide optimum operation for the mission.

7.2 Engine/Vehicle Integration

Performance of ejector/rocket propulsion system is greatly influenced by the amount and condition of ingested air. However Vehicle/engine configuration and geometry is critical in providing the required air to the ejector and in proper mixing of ingested air with rocket engines exhaust flow. It is desirable to use one ejector shroud around cluster of rocket engines rather than one ejector for each engine in order to reduce ejector length and weight and increase mixing. This also requires proper integration of engine with vehicle.

Engine/vehicle integration even thought is a major issue for all rocket systems, it is more critical for air-augmented systems such as ejector/rocket and requires substantial and detailed investigation.
It also must be noted that the size (volume, configuration, and weight) of an ejector/rocket system could impact lift off facility and ground operation/lift-off preparation considerably. Proper coordination with all groups involved is required in order to design an optimized ejector/rocket configuration.

7.3 Air Induction System

For a simple ejector-rocket at low Mach numbers, the performance of secondary air inlet system is not as critical as it is for the range of supersonic speeds when the ingested air is decelerated to subsonic speed by means of shocks. In this case, high total pressure recovery with minimum drag is desirable. While these objectives are certainly emphasized in any air breathing propulsion systems, the overall performance of an air-augmented rocket is not quite as sensitive to these parameters as the performance of a pure air breather such as ramjet.

At low speeds the pumping capability of this system mainly depends on air inlet geometry, ambient and nozzle exhaust flow conditions and shroud geometry. Since most rocket engine nozzle exhaust flow is over-expanded at low speed, primary/secondary flow interaction is complicated by embedded shocks and Mach disks. For ALS type trajectory, the nozzle exhaust flow is over-expanded up to about 26000 ft.

7.4 Mixing/Combustion

The key in achieving high performance is the mixing of primary and secondary flow with minimal loss (entropy rise or total pressure loss). Efficient mixing process is essential and requires efficient momentum exchange between the streams to increase total pressure of secondary flow and to be able to combust the fuel (either excess from exhaust or injected). The mixing and pumping characteristics of nozzle after-burning are dependent on geometric design and operating conditions. The actual exit area may be the most important parameter for controlling inlet-ejector matching. Both mixing and pumping can be altered by the area ratio and shroud L/D (length/diameter). The mixing and pumping characteristics are interdependent, the mixing characteristics cannot be changed without a change occurring in the pumping characteristics. Mixing aids such as vortex generators can enhance mixing and if additional fuel is injected the required shroud length can be reduces.

7.5 Drag

To design a viable system, consideration must be taken in minimizing the overall drag of system including ram-drag and external/internal aerodynamic drag. It is obvious that thrust augmentation could only be realized if static pressure of mixed/burned mixture exceeds ambient pressure.

7.6 Boundary Layer Effects

Boundary layers developed on rocket nozzle wall and secondary air induction system will affect system's pumping capability, momentum exchange between the two streams and total pressure of mixed region. Flow separation due to adverse pressure gradient in the boundary layer and shock/boundary layer interaction will influence ejector flow and performance.

Therefore, in the design process the effects of boundary layer and possibility of boundary layer bleed system must be considered.

7.7 Cooling

Due to high temperature of mixed and combusted secondary air, special cooling consideration might be appropriate. In the air induction system, some cooling might be required for leading edges and local high heat flux area due to shock impingement and shock boundary layer interaction.

8. RECOMMENDATIONS

Limited amount of effort is required to finalize the potential gains calculated with injection of additional fuel. The calculated gains did not represent the actual potential gains achievable, with the ejector geometry used, since performance/trajectory mismatch was observed with the level of iteration (1 iteration) performed. Therefore the potential gains with fuel addition should be properly determined with the baseline ejector geometry with additional iterations to match performance to flight trajectory. In addition, since thrust augmentation with injection of additional fuel was based on ejector geometry optimized with parametric study performed for the cases without fuel addition, an optimum ejector design for fuel addition cases needs to be identified and its maximum potential gains needs to be calculated with matched trajectory/performance.

The simplified assumptions made to perform this study were stated previously. In order to eliminate uncertainties in the obtained results a rough magnitude effects of these simplifications, as a minimum effort, must be determined. More elaborate, still simplified, analyses are required to estimate the level of losses such as mixing, internal drag, wall heat transfer, flow non-uniformities, chemical kinetics and shock losses. The effect of shroud design on ejector flow, inlet performance and secondary air flow must also be investigated and to assure entrainment (ingestion) of proper/sufficient amount of air. Advanced analysis techniques such as Computational Fluid Dynamic (CFD) can be used to characterize ejector flow field in order to determine the ejector losses accurately and to identify any flow irregularities in the ejector (such as separation). In addition, the external flow around the vehicle can also be modeled and included in the CFD effort to obtain a better understanding of effect of vehicle size and geometry on inlet and ejector operation.

Following the completion of the analysis and selection of the ejector geometry (design), a component test series, including hot-fire tests, is necessary to verify feasibility and viability of the concept. Major issues such as pumping capability, mixing, and level of ejector performance (thrust) should be evaluated. Component testing would provide an opportunity to verify proper operation of the ejector/rocket system under realistic conditions and to confirm the results from the analysis.

The tests could be performed at Rockwell International test facilities such as Advanced Propulsion Test Facility (APTF) at Rocketdyne's Santa Susana Field Laboratory (SSFL) for static tests and North American Aircraft's (NAA) Tri-sonic Tunnel for low speed flight regime tests. The existing Rocketdyne's rocket engine hardwares (thrust chambers) can be utilized with minor hardware modifications (installation of the ejector) in order to minimize the cost and hardware fabrication and test schedule.

As a minimum, ejector mass flow, wall pressure and wall temperature should be measured. The capability exits to survey ejector flow field (species concentration) and to measure engine thrust level, flow velocity, temperature, and total pressure. The effects of mixing aids, such as axial vortex generators, on the mixing and ejector length can be also investigated during testing.

It is also recommended to investigate similar/alternate concepts and assess their effectiveness and viability. A variable geometry ejector design can be considered to increase the effectiveness and/or to extend the range of operation. A variable geometry design with added complexity will result in more efficient air induction system up to about flight Mach 5-6 and can result in higher gains. With minimal moving parts (like inlet door(s)) an ejector can also be utilized as an extension of rocket engine nozzle to increase thrust at higher altitude operation while at the same time with lower area ratio nozzle, weight can be reduced with increased performance at lower altitudes.

The potential gain achieved with ideal flow calculation shows that ejector/rocket system is equivalent to a all rocket system having increased Isp of about 40 sec., which indicates that ejector/rocket system has great potential for Single Stage To Orbit (SSTO) application. With the high performance level achived, ejector/rocket is a great alternative to all rocket system for earth to orbit missions, and has significantly reduced design complexity and required technology development compare to other combined cycle propulsion systems for SSTO.

9. CONCLUSIONS

This preliminary study results show significant ideal potential gains with Rocket Engine Nozzle After-burning (ejector/rocket), for ALS type missions, exist with properly designed ejector shroud. Rocket thrust augmentation was achieved with fixed size ejector with and without additional injected fuel for low speed regime (flight Mach 0 to 2). The calculated thrust augmentations obtained with ejectors were traded off against shroud weight and additional external aerodynamic drag and resulted in estimated 27% increase in ALS payload with fixed ALS baseline vehicle size. Based on sensitivity factors (partial derivatives) developed for the ALS baseline vehicle with about 120,000 lbs. payload, an increase in engine performance of DI_{sp}=40 secs. would be required to achieve the same payload increase. 19% and 22% reduction in ALS baseline Gross Lift-Off Weight (GLOW) and propellant weight were also estimated with ALS fixed payload of about 120 Klbs.

The results of this preliminary study with its limited scope indicate that a properly designed ejector rocket system is a viable concept with high potential pay-offs. However, uncertainties (such as pumping capability at low speed, primary/secondary flow mixing, interaction effects and ejector length, effect of flow non-uniformities and boundary layer, level of losses) from adopting simplified approach and assumptions in this study, in regards to effectiveness of an ejector/rocket system raise some concerns that can only be resolved and answered by further thorough investigations.

Shahram Farhangi

Shahram Farhangi
Advanced Combustion Devices Analysis

ROCKET ENGINE NOZZLE AFTER-BURNING CONCEPT

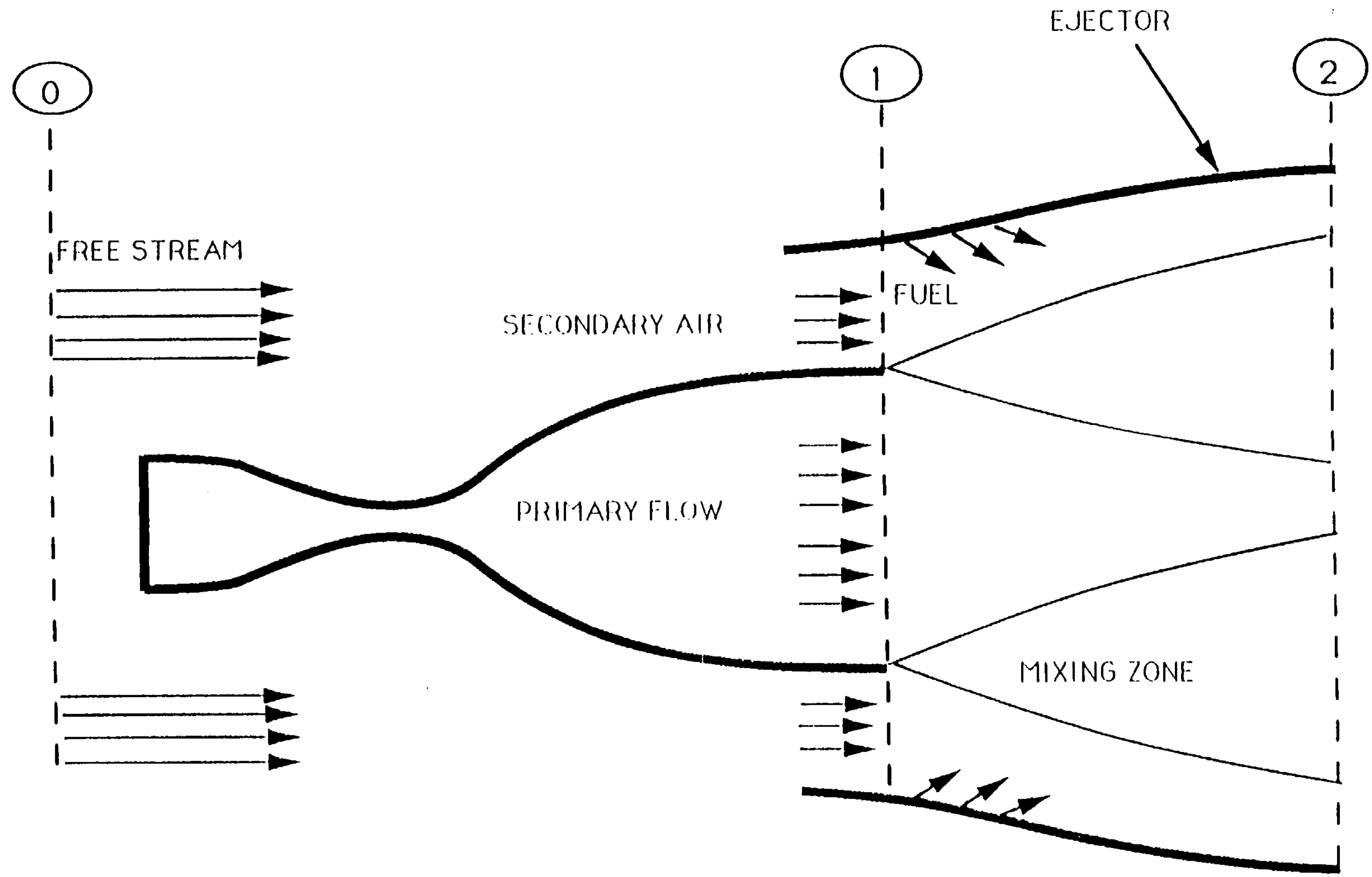


FIGURE 1.

Ejector weight varies linearly with As

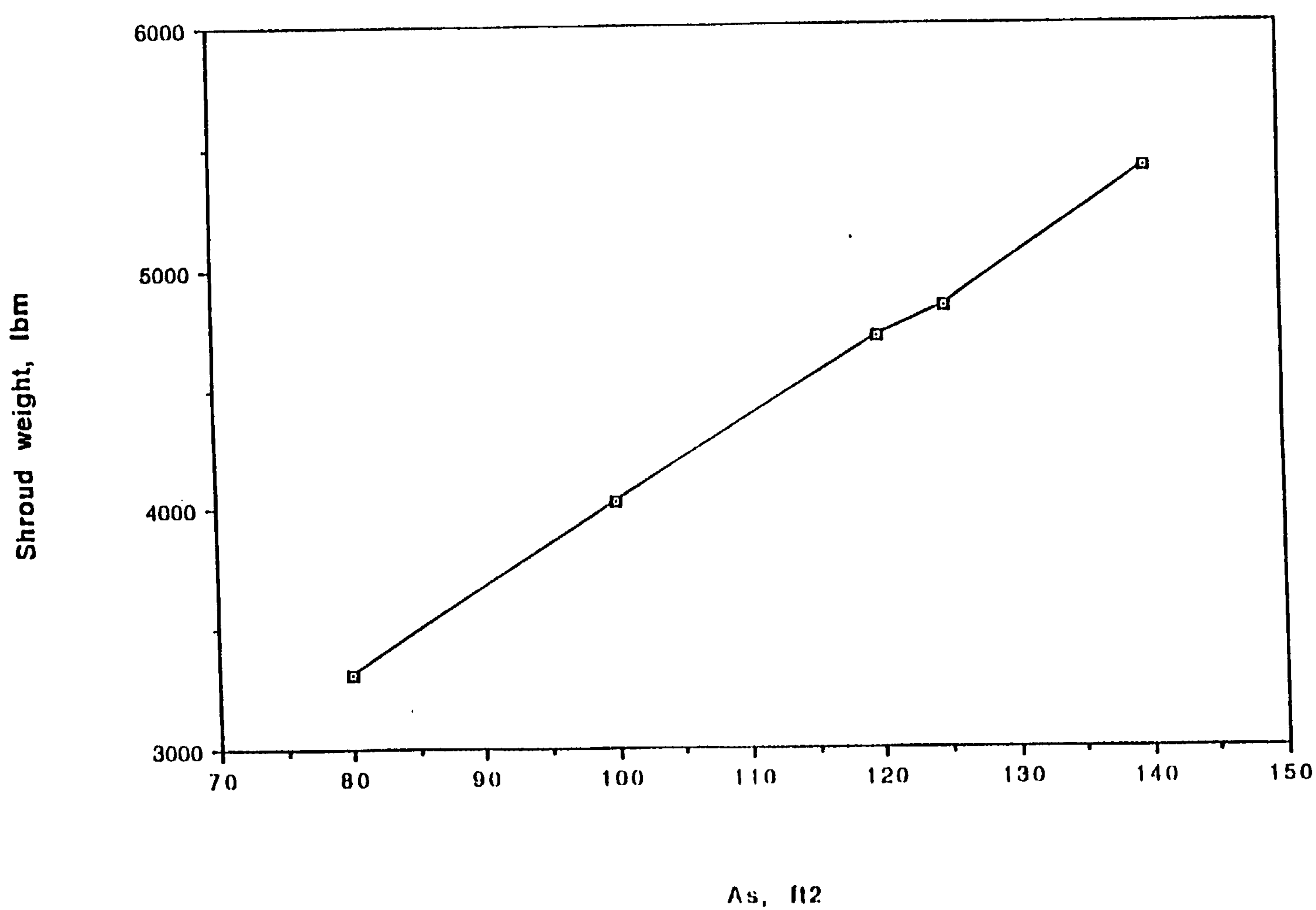


FIGURE 2.

Augmentation VS Ejector Area Ratio

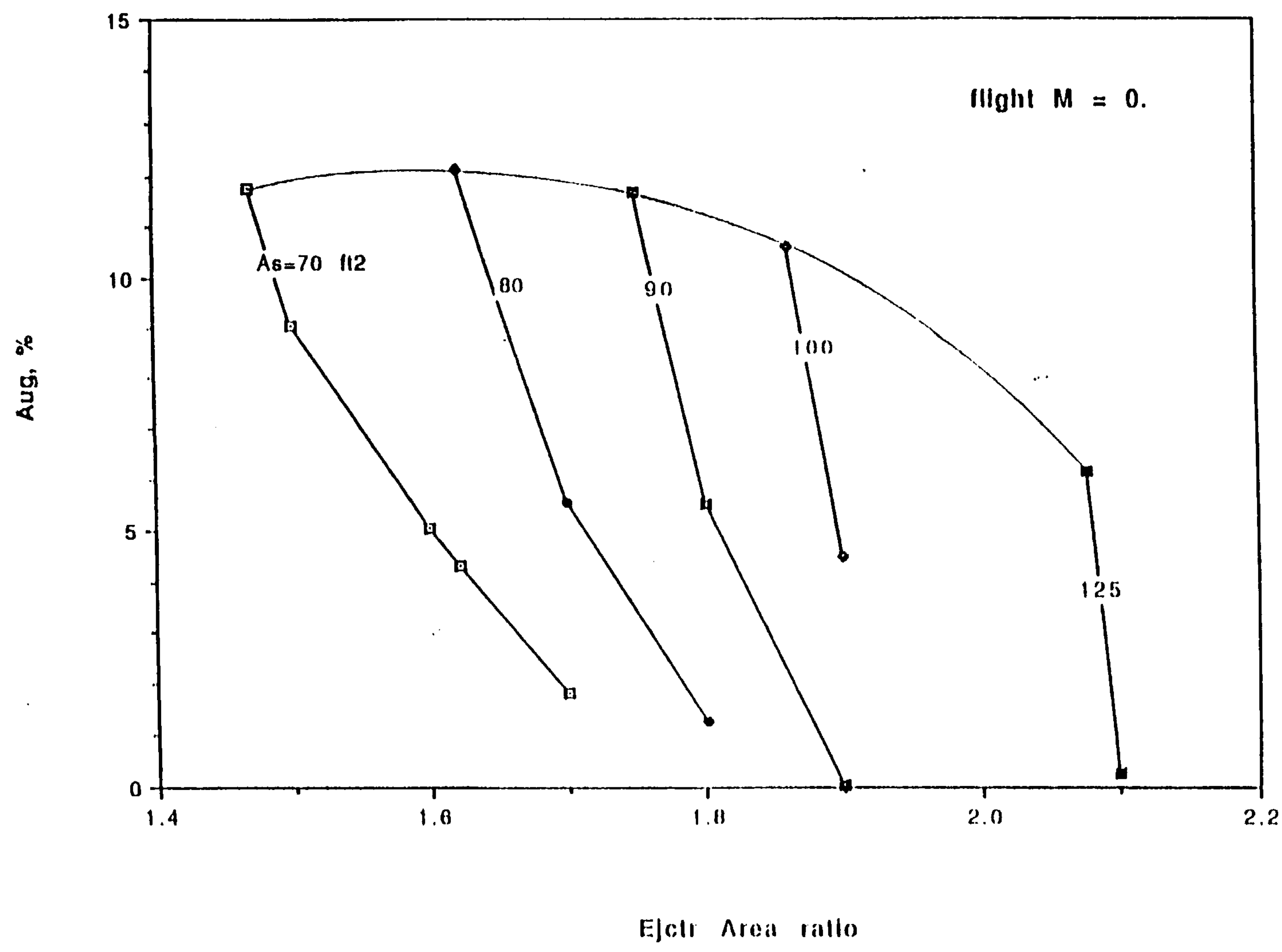


FIGURE 3.

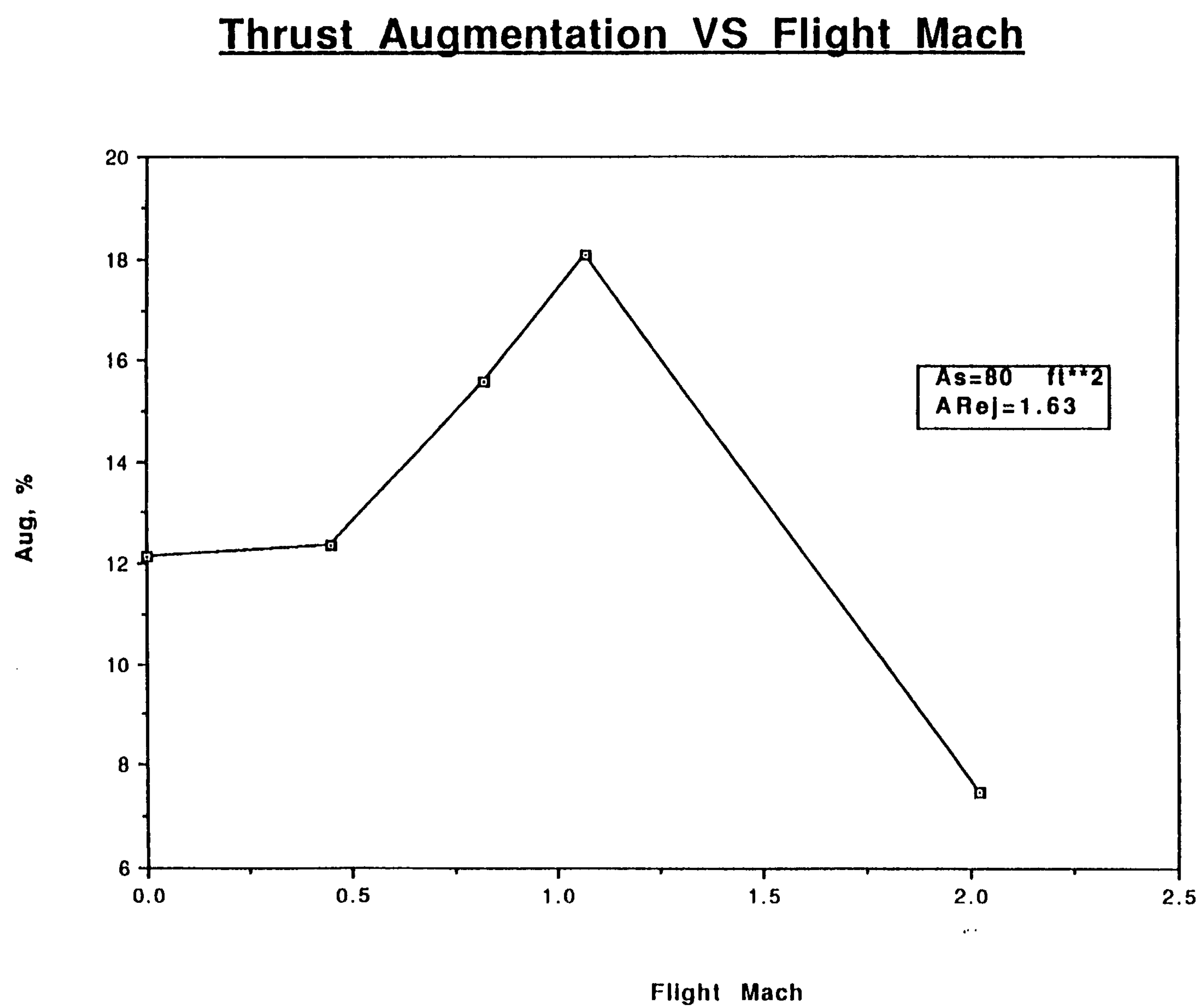

Thrust Augmentation VS Flight Mach
As=80 ft**2
ARej=1.63
Aug, %
Flight Mach
20
18
16
14
12
10
8
6
0.0
0.5
1.0
1.5
2.0
2.5

FIGURE 4.

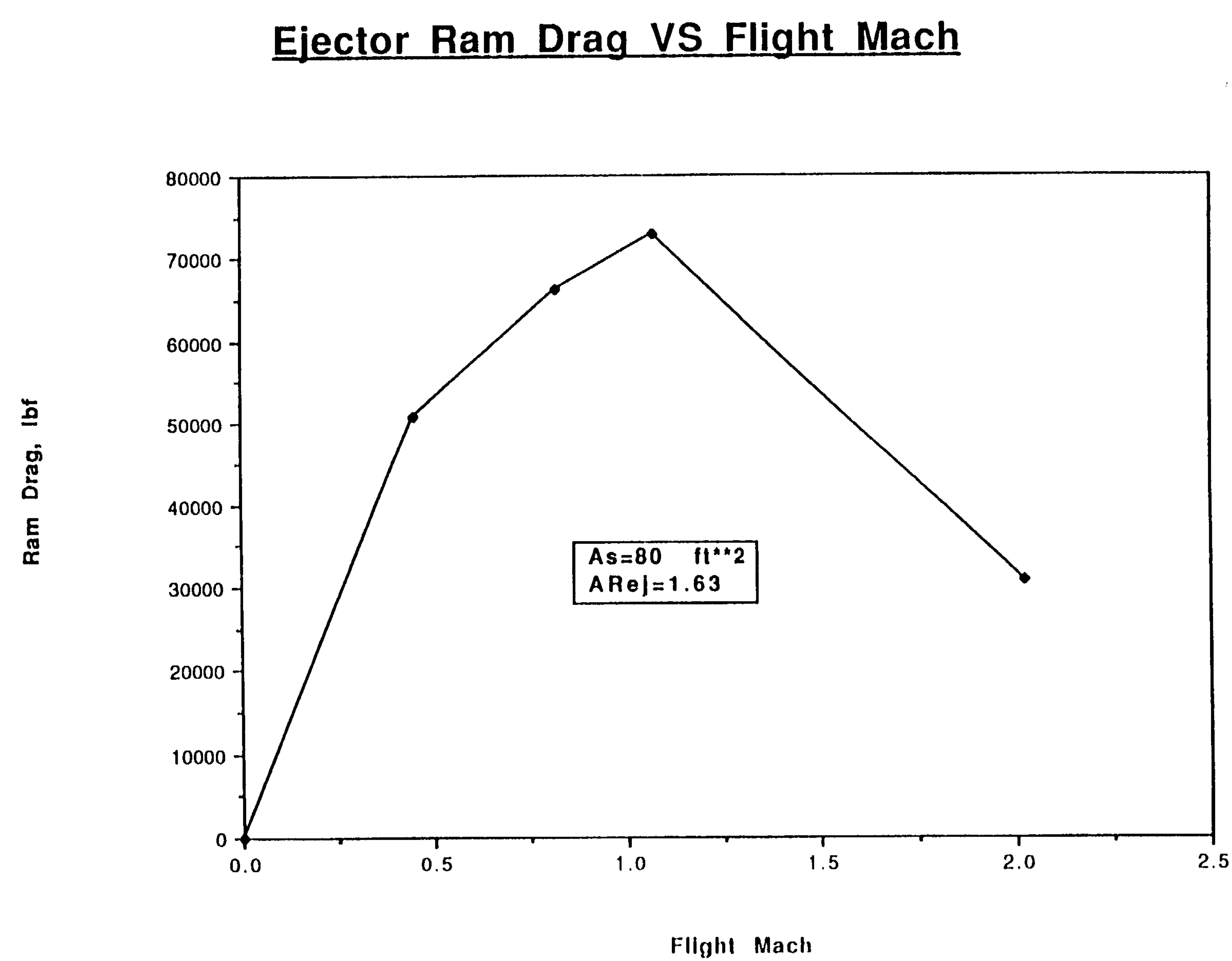

Ejector Ram Drag VS Flight Mach
Ram Drag, lbf
80000
70000
60000
50000
40000
30000
20000
10000
0
0.0
0.5
1.0
1.5
2.0
2.5
As=80 ft**2
ARej=1.63
Flight Mach

FIGURE 5.

Mass Flow Ratio vs Flight Mach

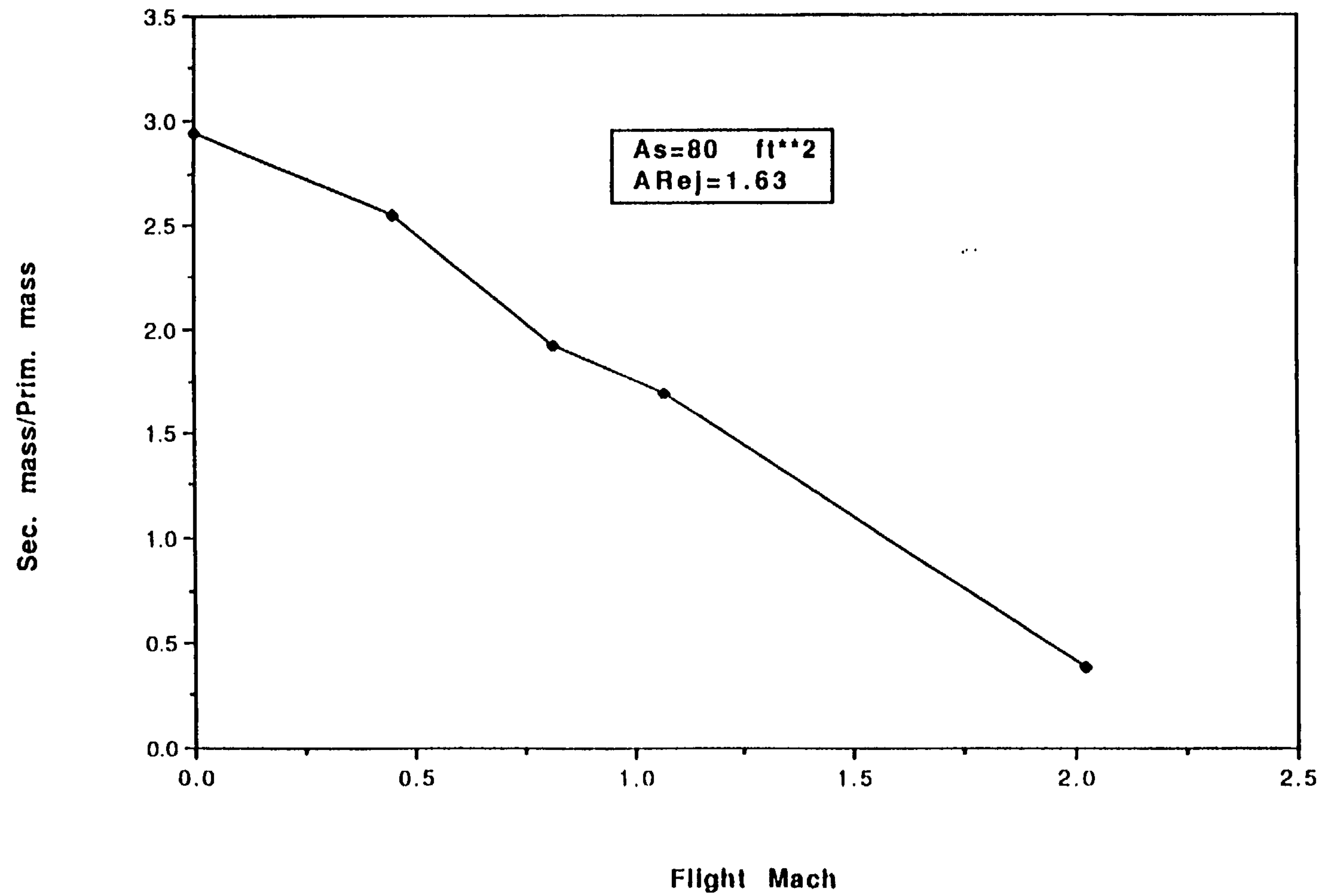


FIGURE 6.

Augmentation VS Ejector Area Ratio

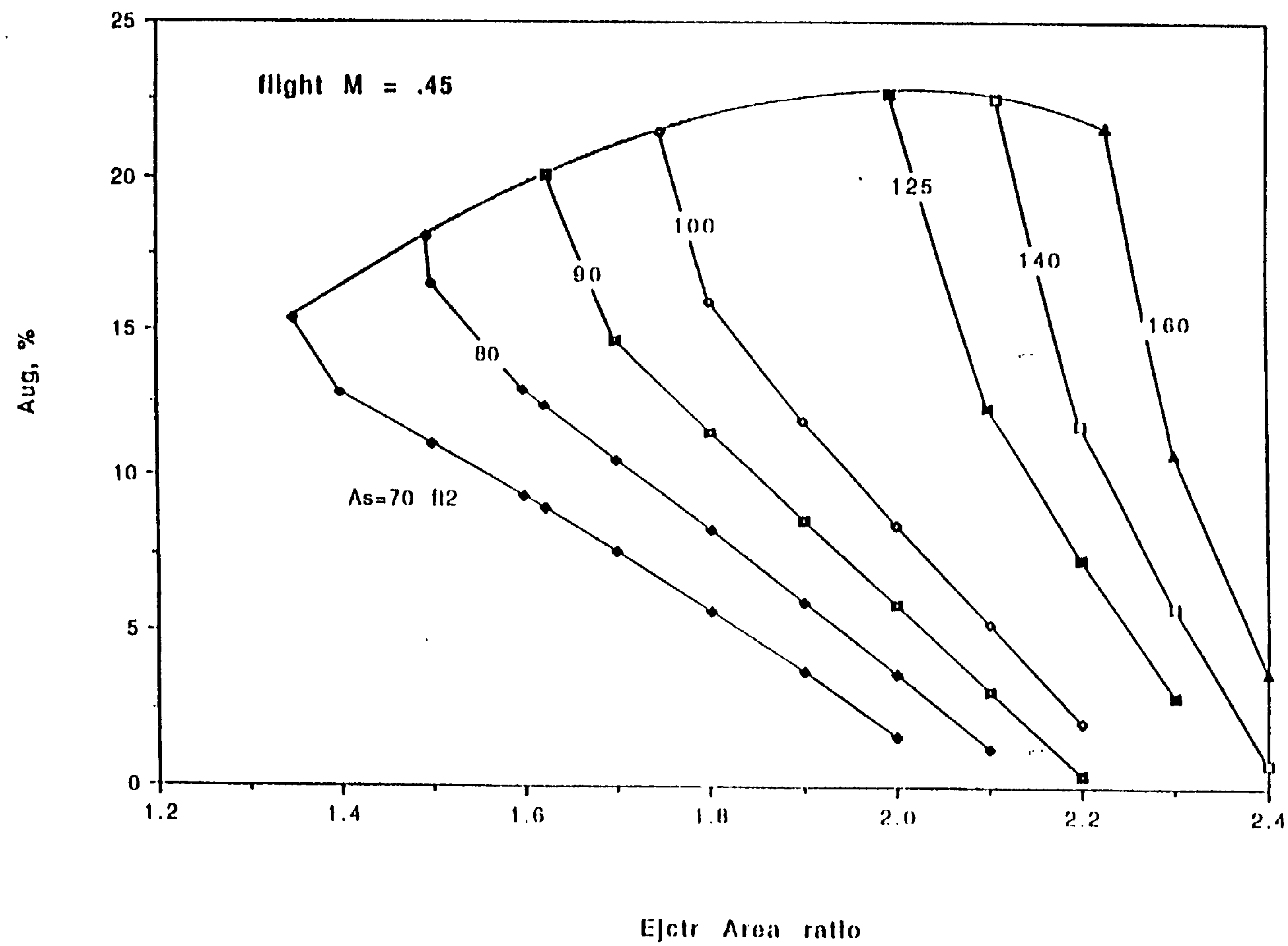


FIGURE 7.

Thrust Augmentation VS Flight Mach

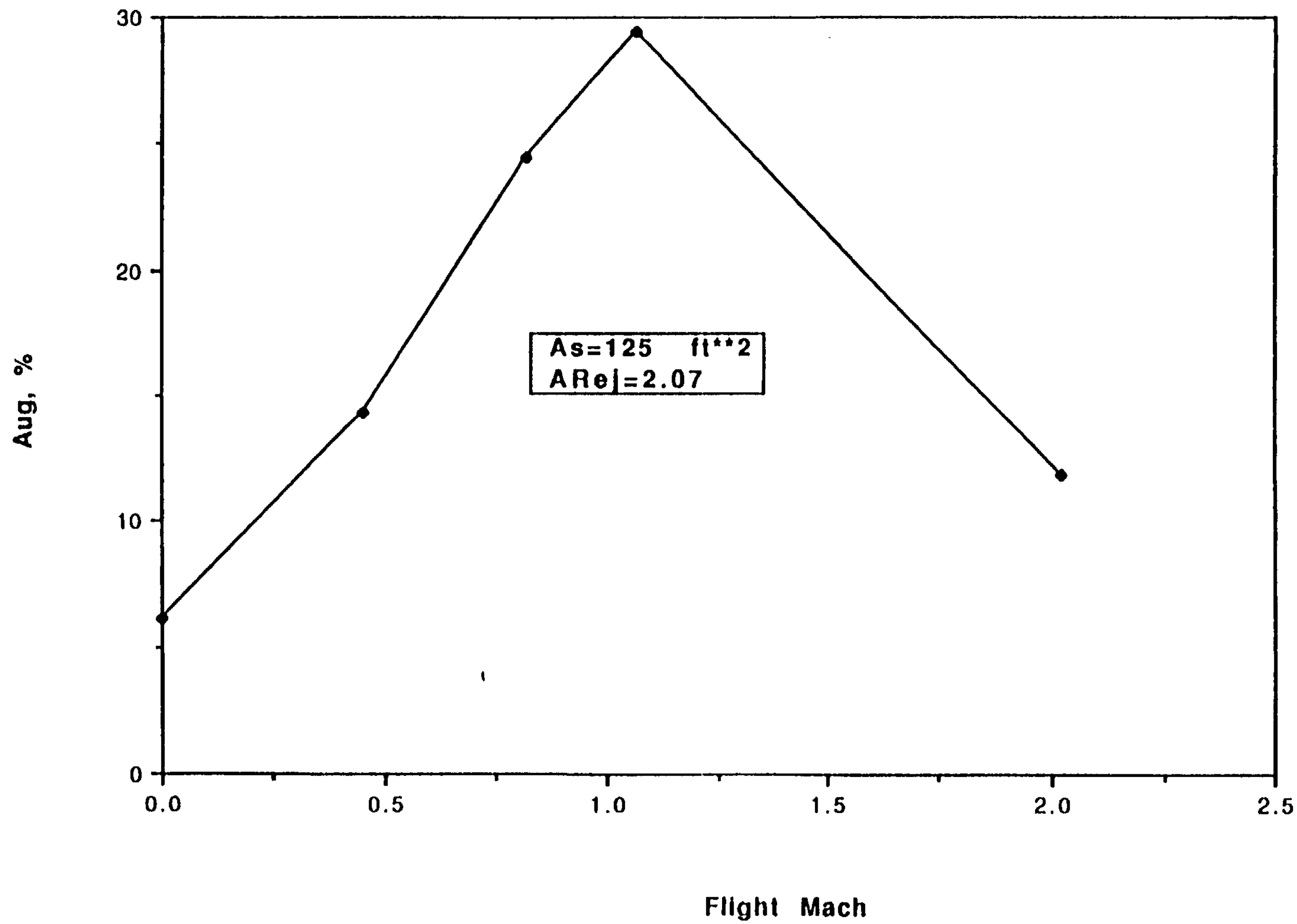


FIGURE 8.

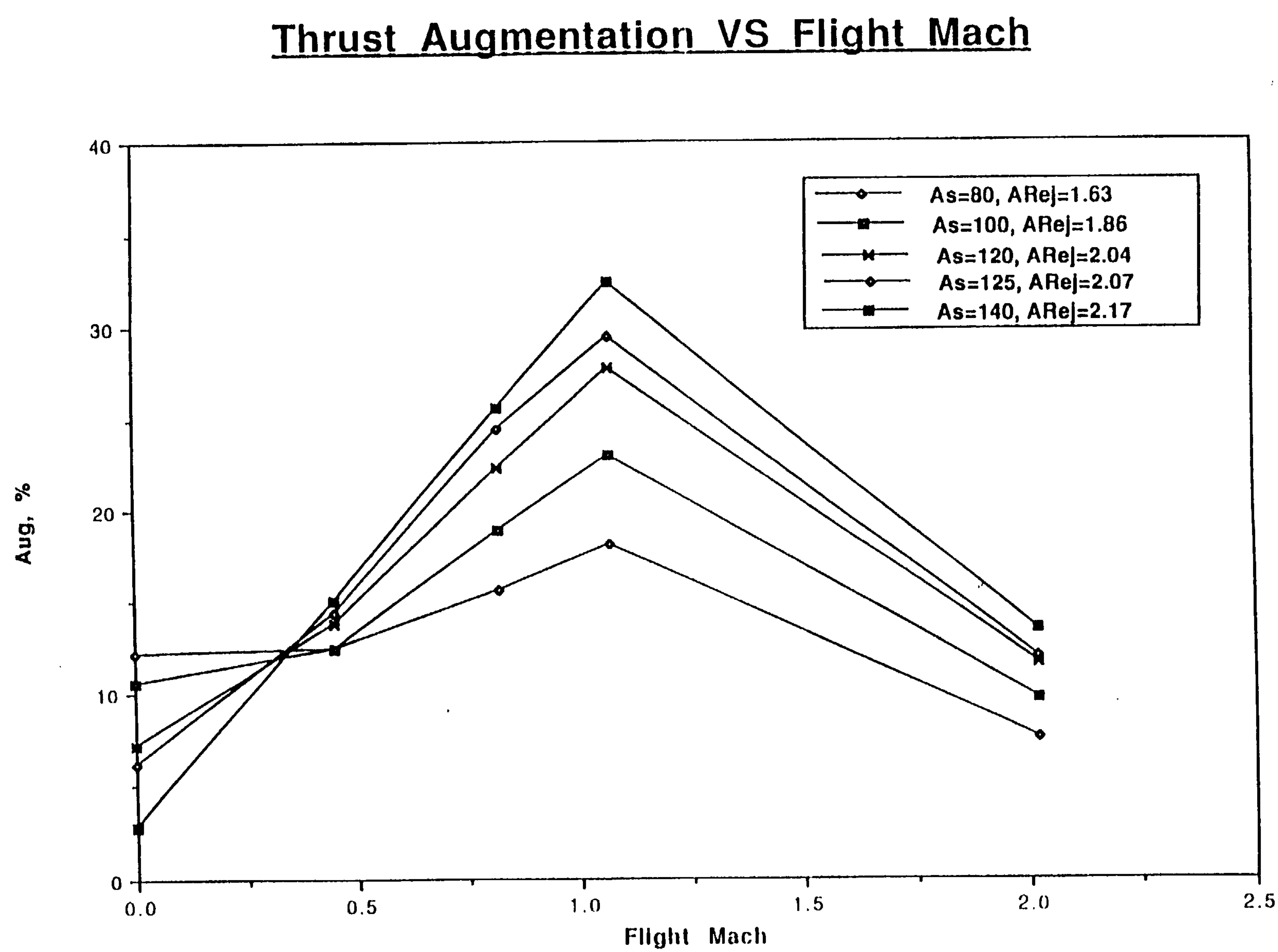

Thrust Augmentation VS Flight Mach
40
30
20
10
0
0.0
0.5
1.0
1.5
2.0
2.5
Aug, %
Flight Mach
As=80, ARej=1.63
As=100, ARej=1.86
As=120, ARej=2.04
As=125, ARej=2.07
As=140, ARej=2.17

FIGURE 9.

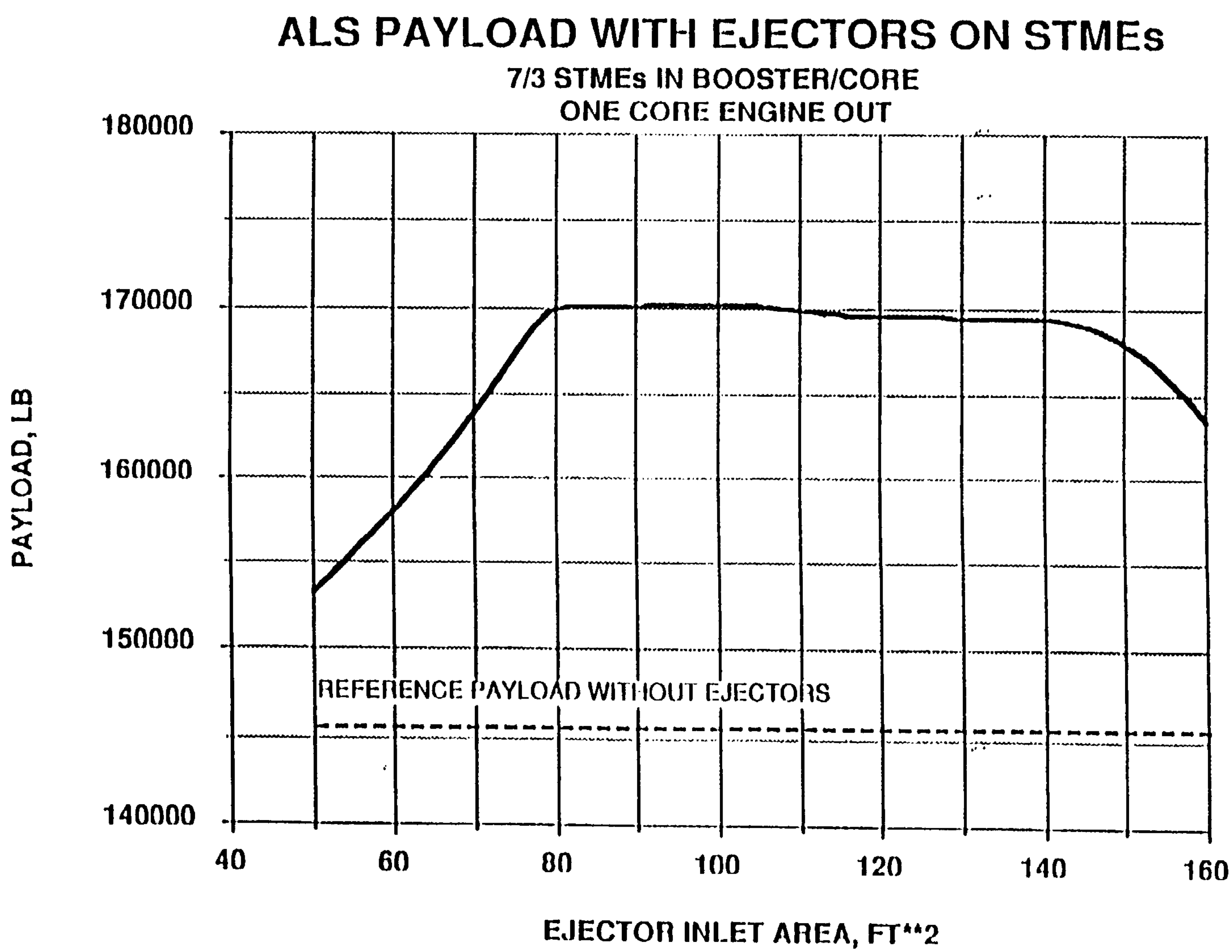

ALS PAYLOAD WITH EJECTORS ON STMEs
7/3 STMEs IN BOOSTER/CORE
ONE CORE ENGINE OUT
PAYLOAD, LB
180000
170000
160000
150000
140000
REFERENCE PAYLOAD WITHOUT EJECTORS
40
60
80
100
120
140
160
EJECTOR INLET AREA, FT**2

FIGURE 10.

Thrust Aug. Comparison W & W/O Added Fuel

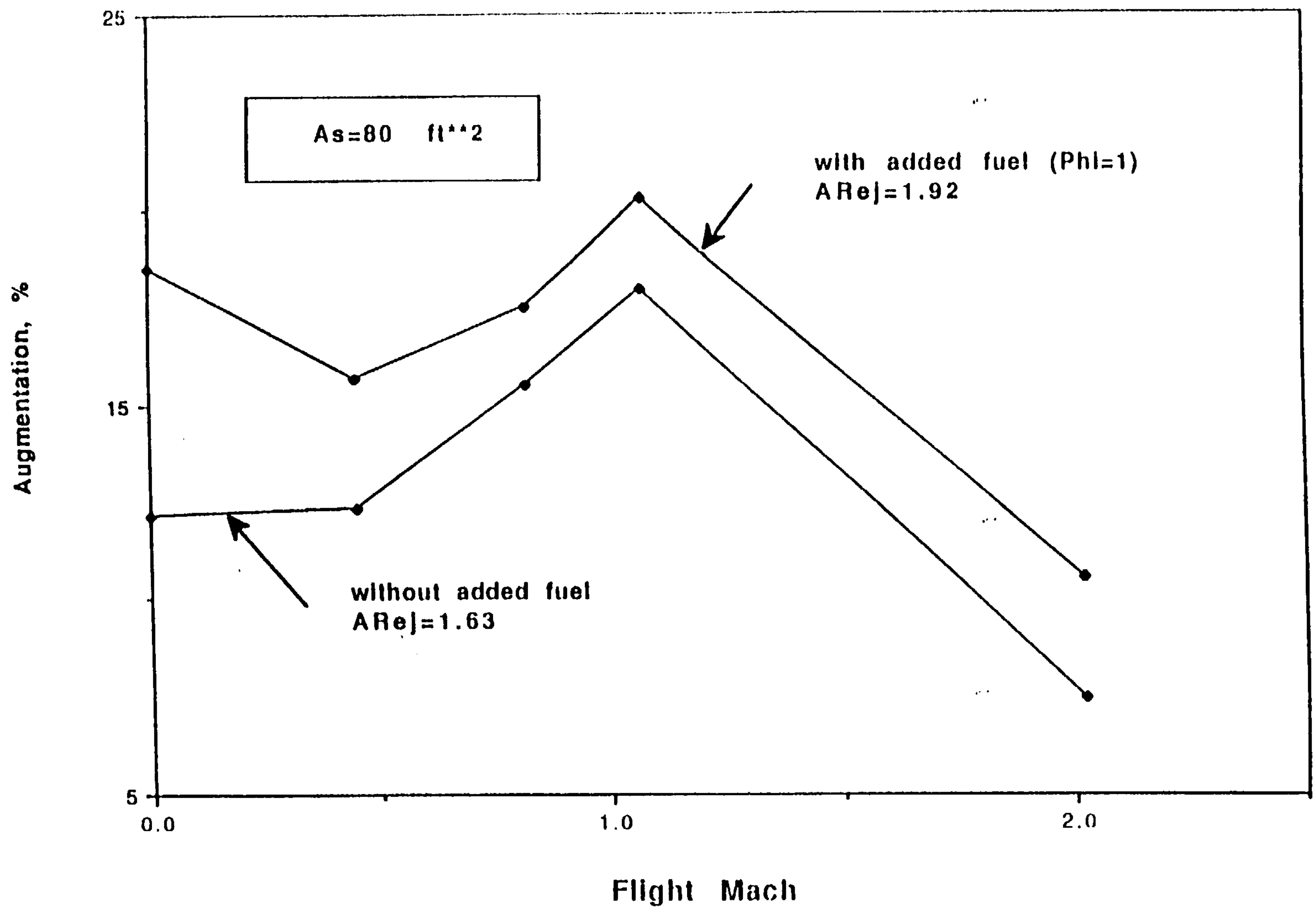


FIGURE 11.

Thrust & Isp Increase With Additional Injected Fuel

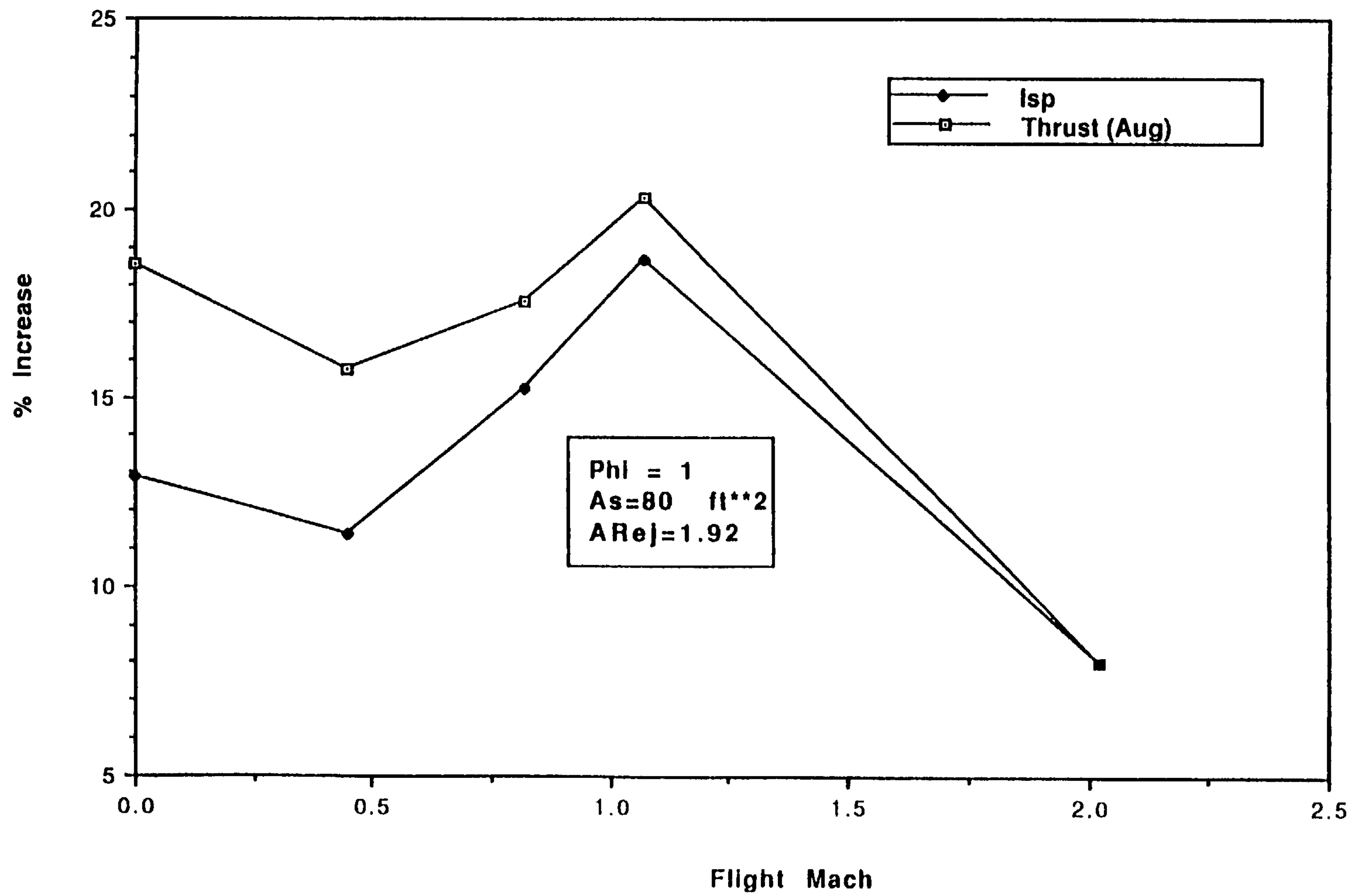
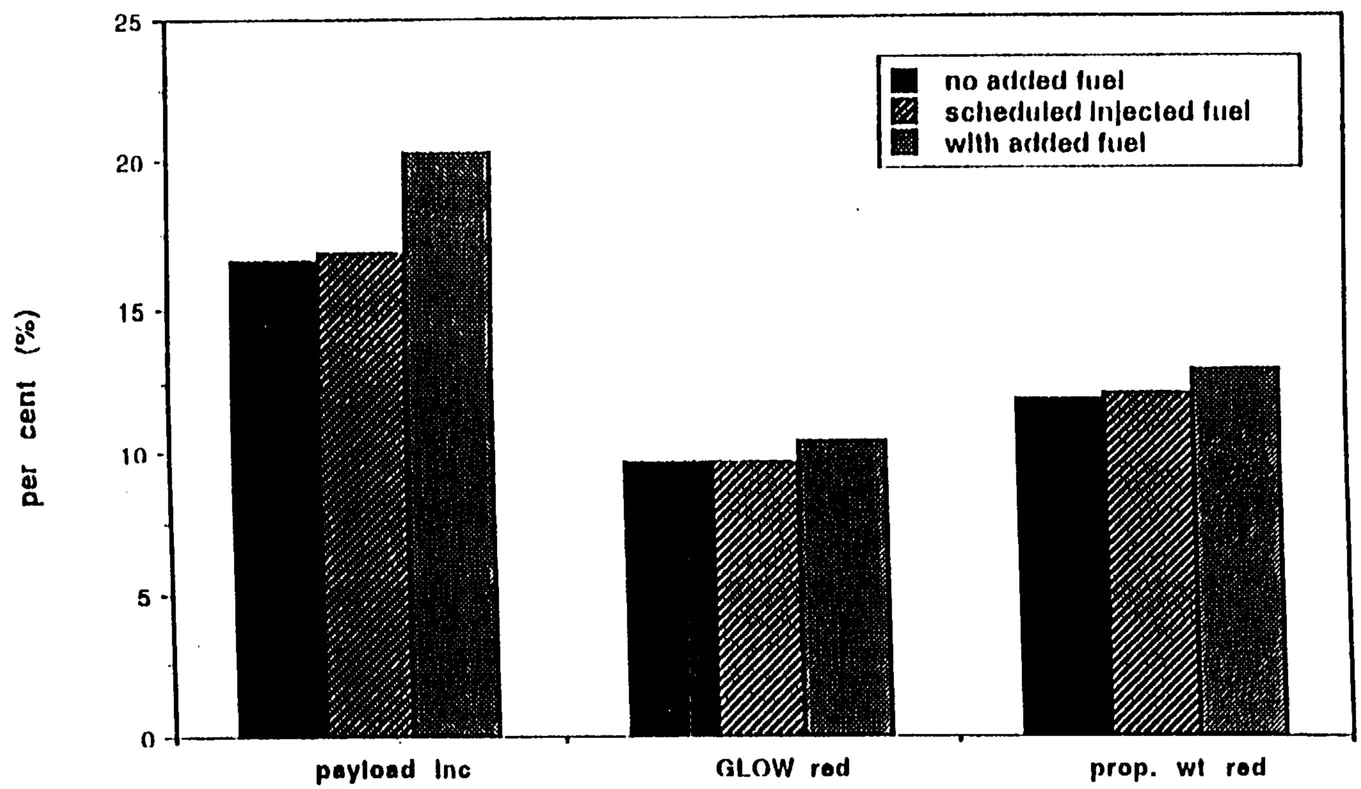


FIGURE 12.

Pronounced Effect With Full Range Fuel Addition



Note: As=80 ft**2, ARej=1.63 & 1.92, Ref P/L=146klb

FIGURE 13.

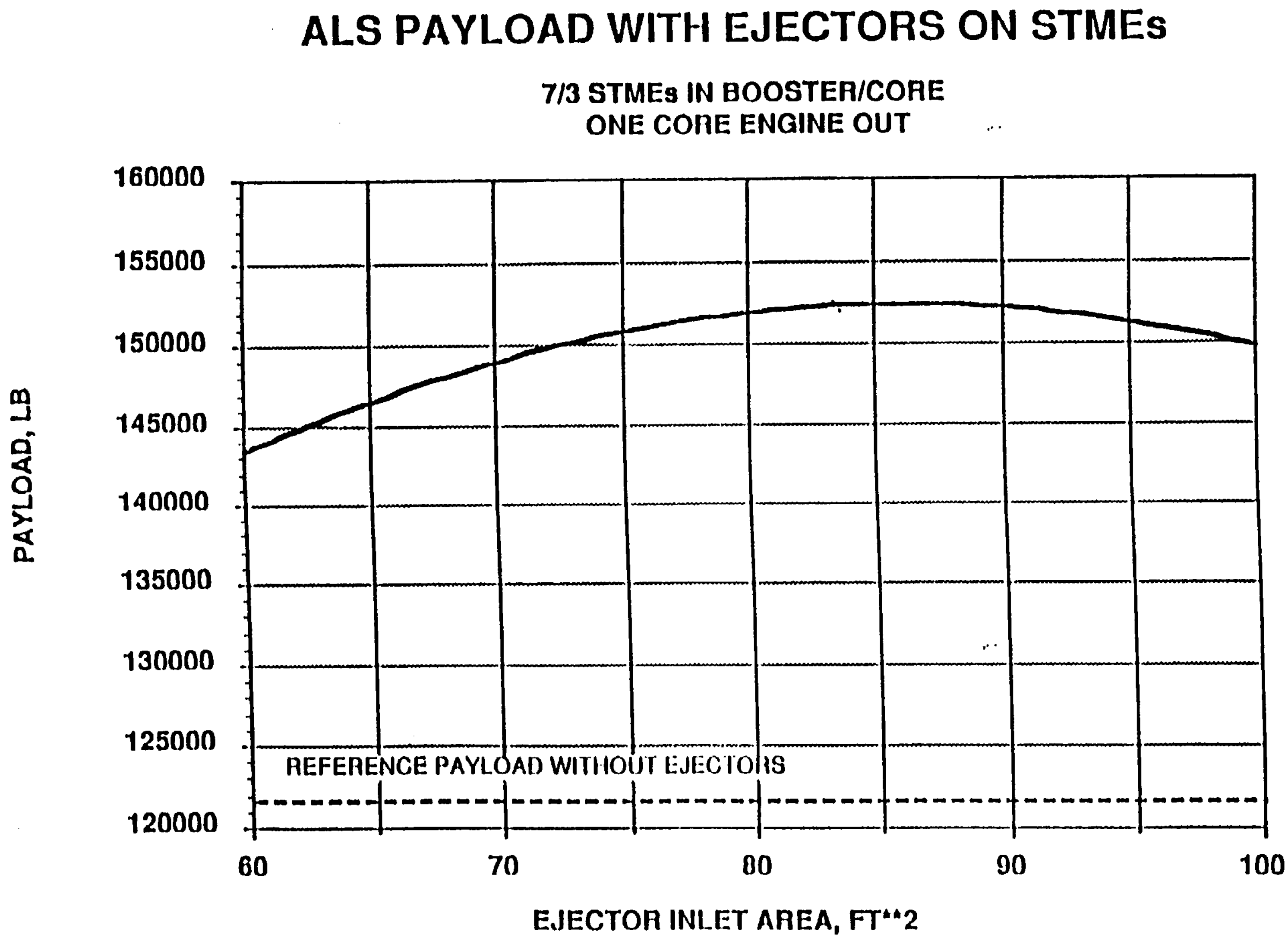
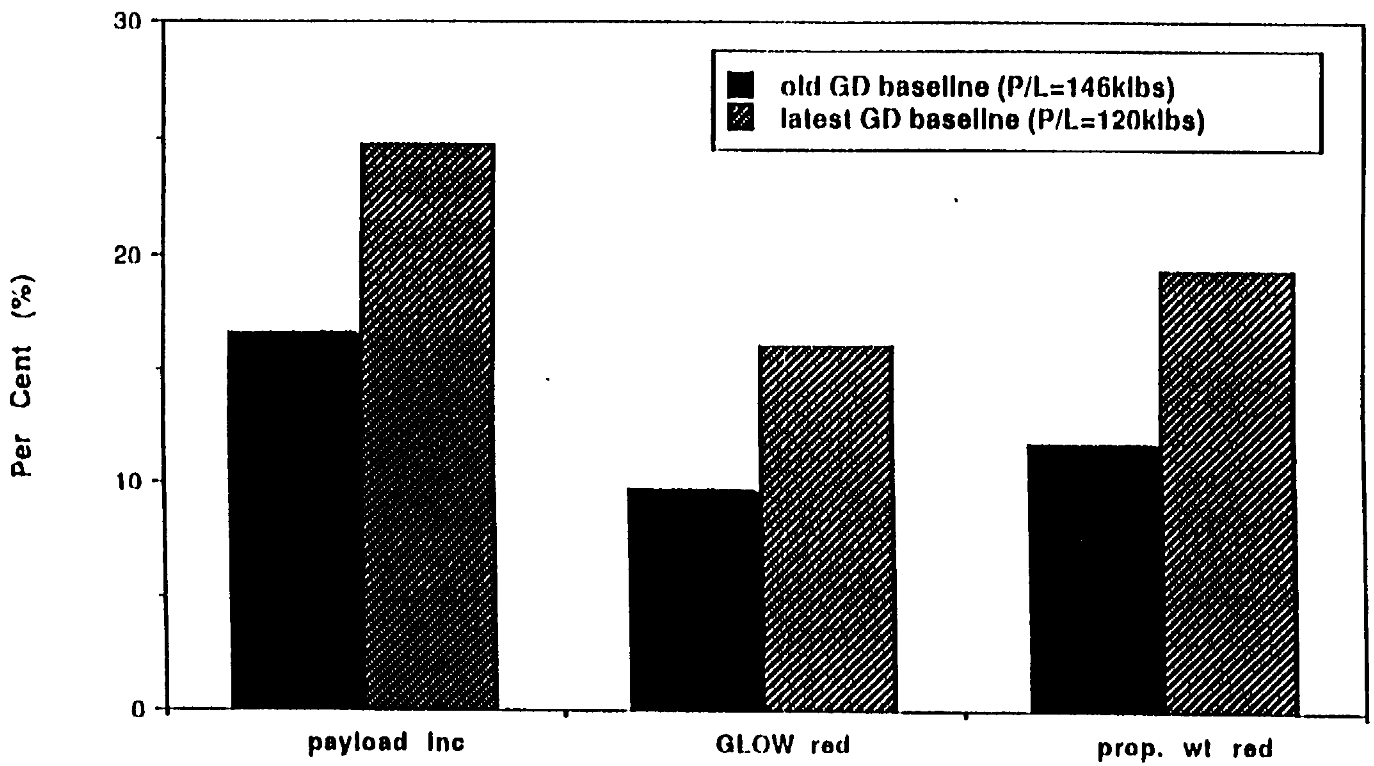

ALS PAYLOAD WITH EJECTORS ON STMEs
7/3 STMEs IN BOOSTER/CORE
ONE CORE ENGINE OUT
160000
155000
150000
145000
140000
135000
130000
125000
120000
PAYLOAD, LB
REFERENCE PAYLOAD WITHOUT EJECTORS
60
70
80
90
100
EJECTOR INLET AREA, FT**2

FIGURE 14.

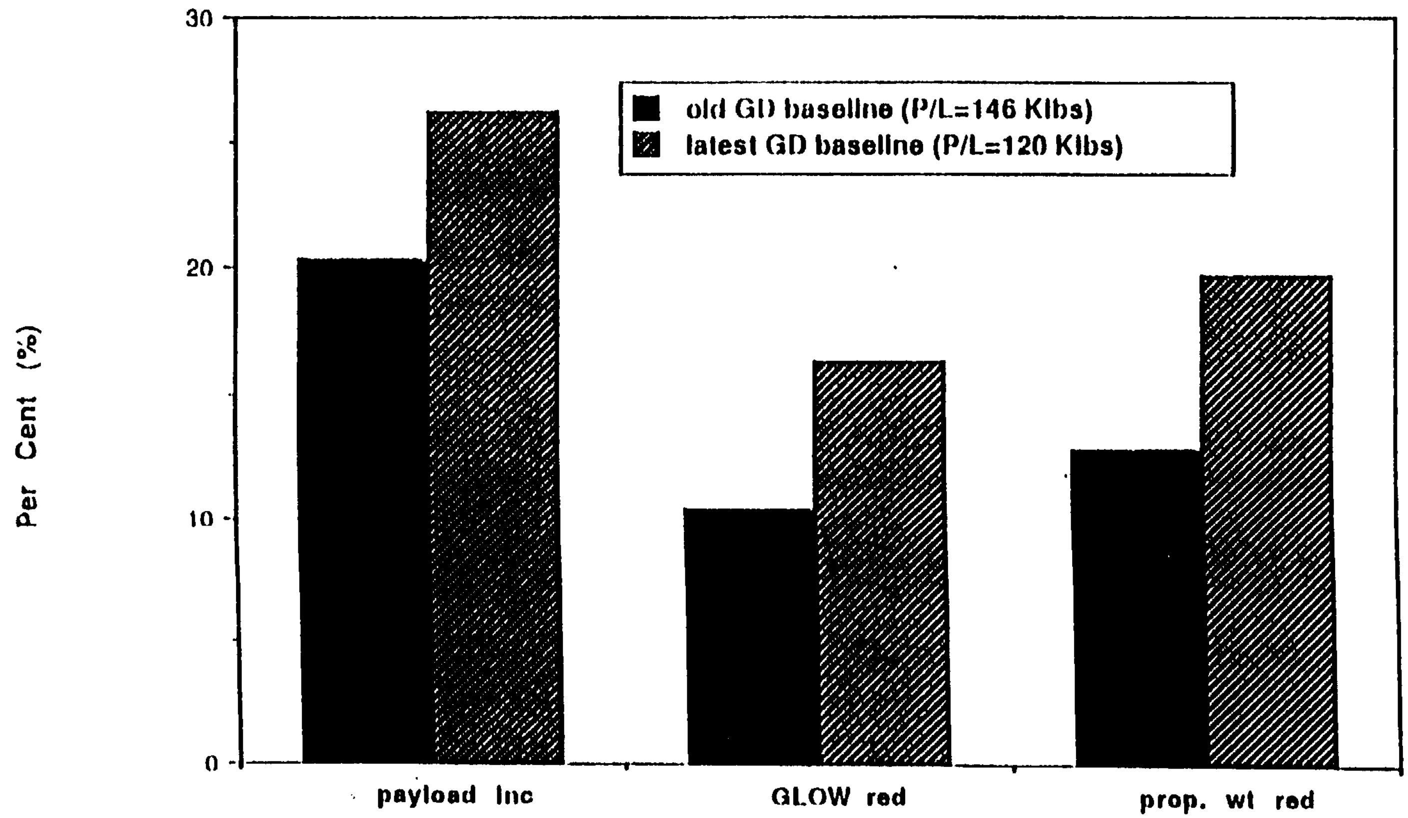
ALS Vehicles With Ejector Installed



Note: As=80 ft**2, ARej=1.63, W/o added fuel

FIGURE 15.

ALS Vehicles With Ejector Installed



Note: As=80 ft**2, ARej=1.92, W added fuel

FIGURE 16.

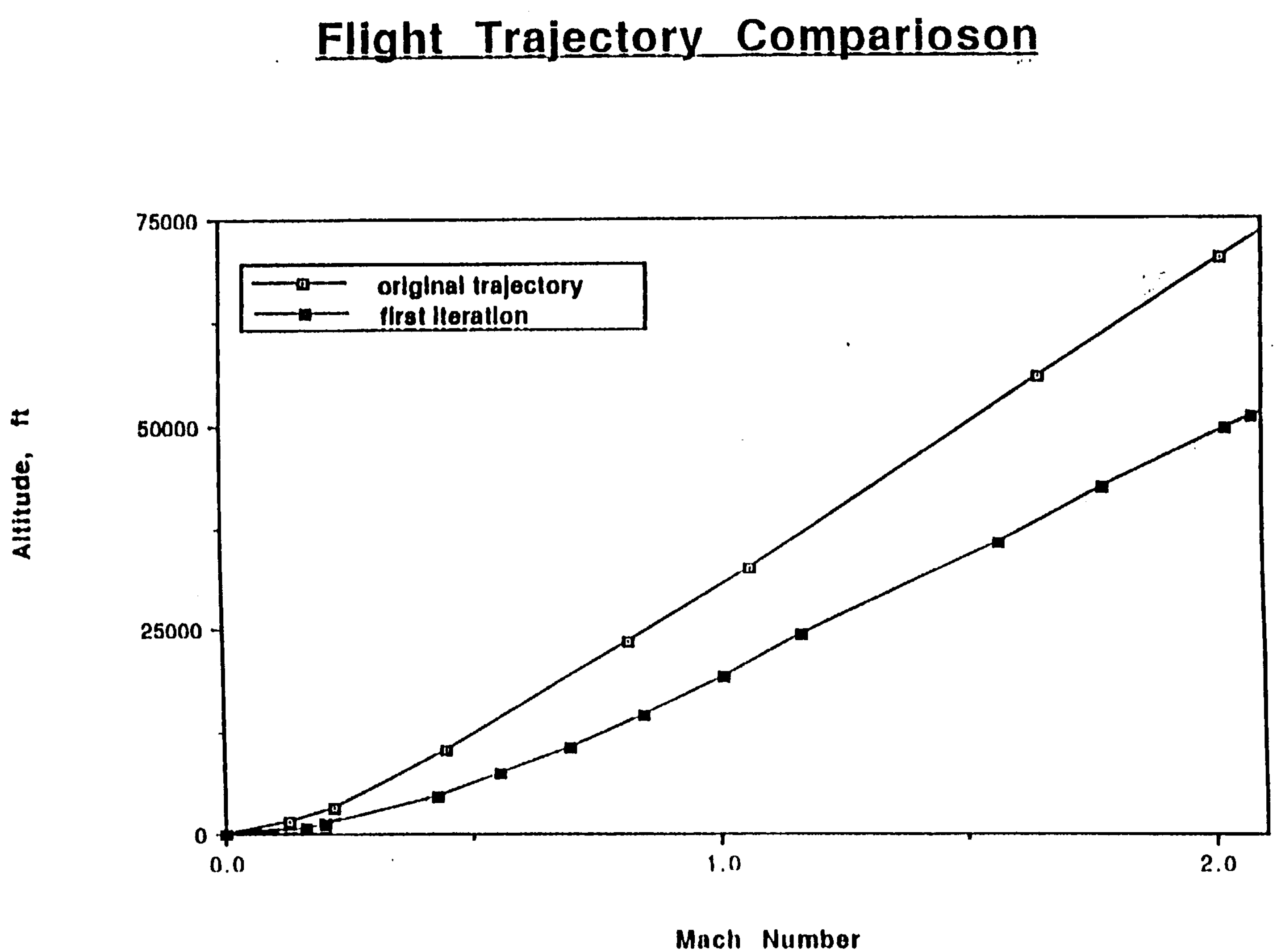

Flight Trajectory Comparioson
75000
50000
25000
0
Altitude, ft
original trajectory
first iteration
0.0
1.0
2.0
Mach Number

FIGURE 17.

Matched Trajectory - Increased Ejector Effectiveness

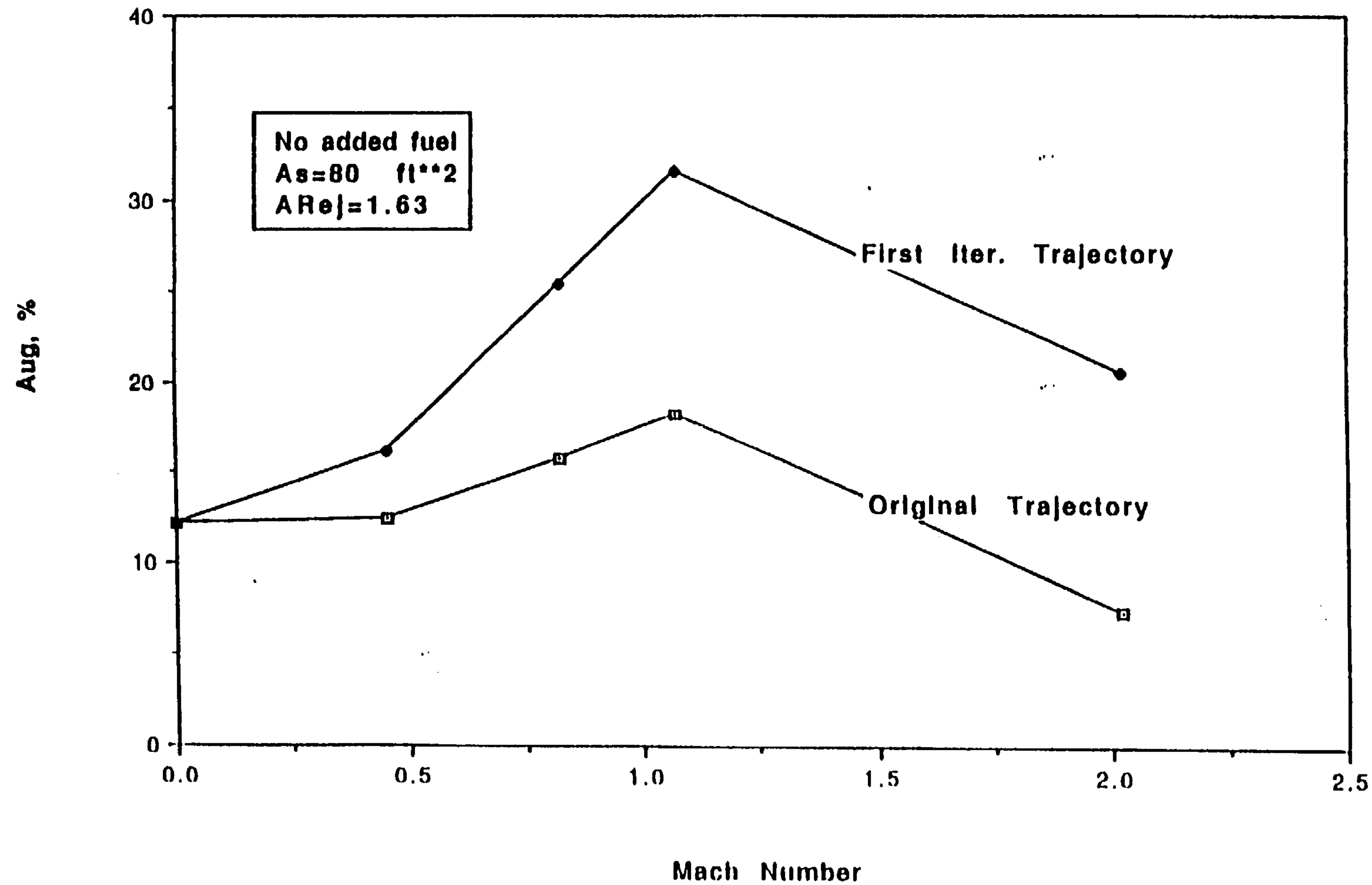
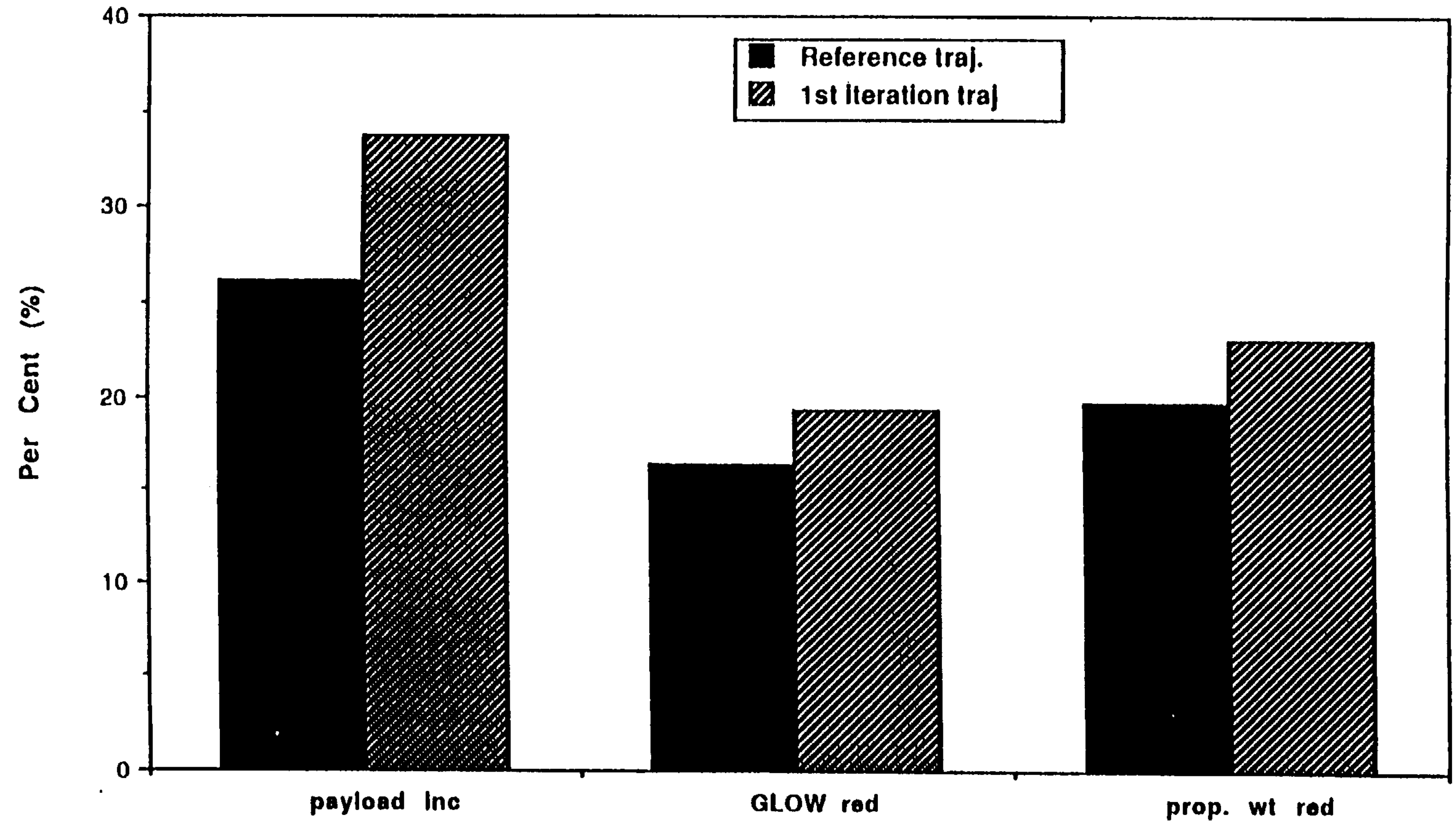


FIGURE 18.

First Traj. Iteartion Gains With Added Fuel



NOTE: As=80 ft**2, ARej=1.92, Ref P/L=120 klbs, Ref traj=1st iter w/o fuel

FIGURE 19.

Flight Trajectroy Comparison

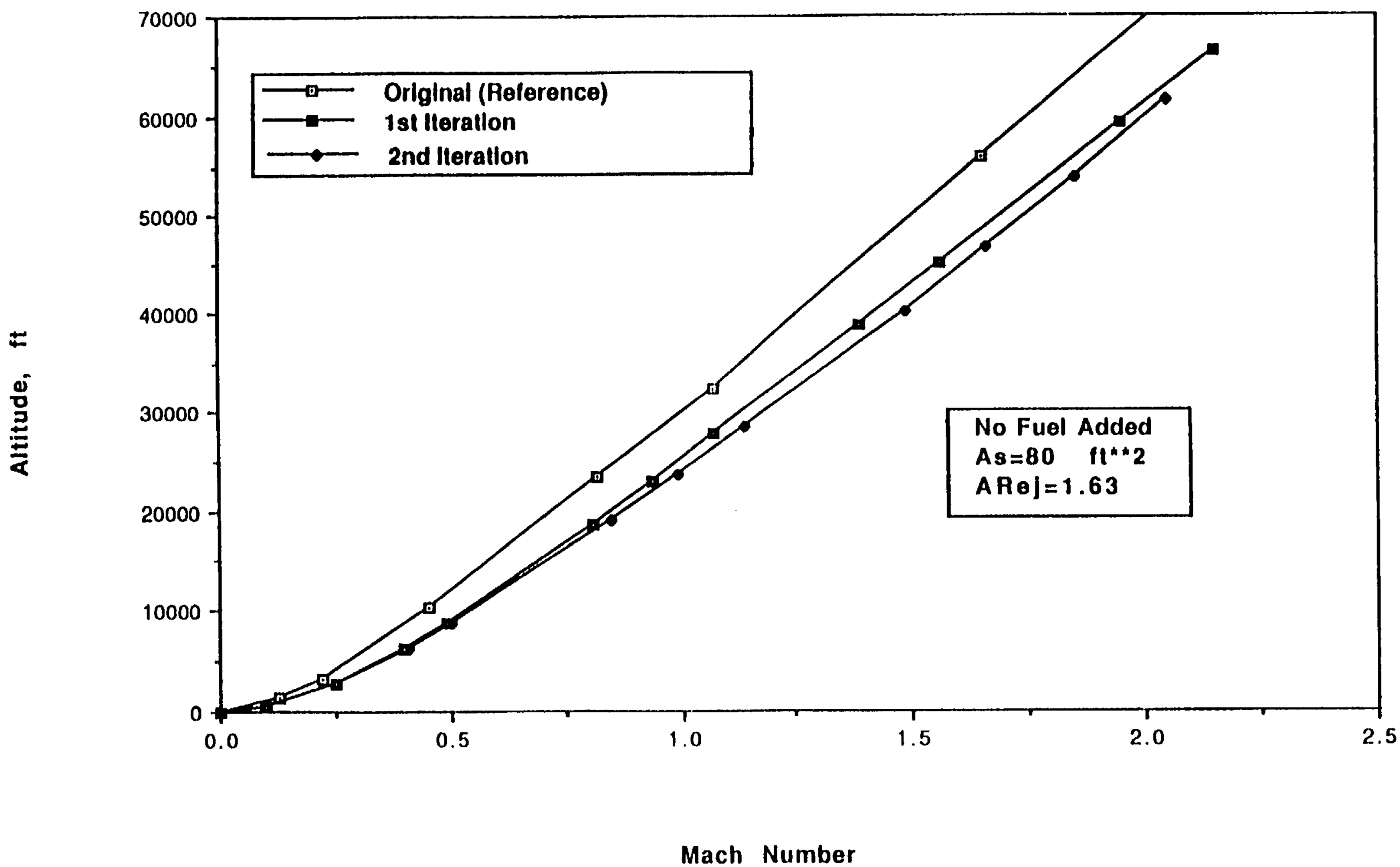


FIGURE 20.

Payload Increase vs Ejector Length

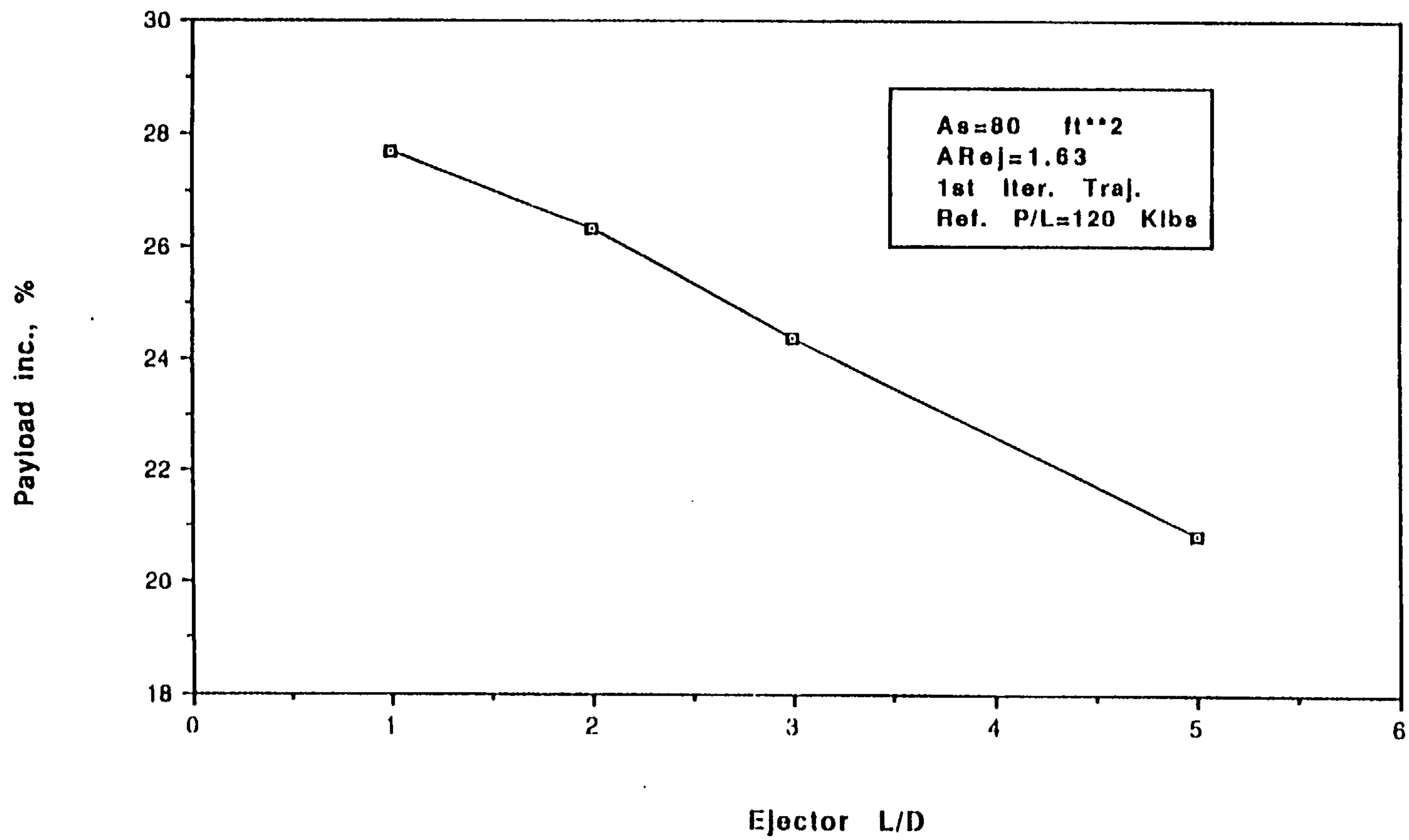


FIGURE 21.

Thrust Aug. Increases With Closer Matched Traj./perf.

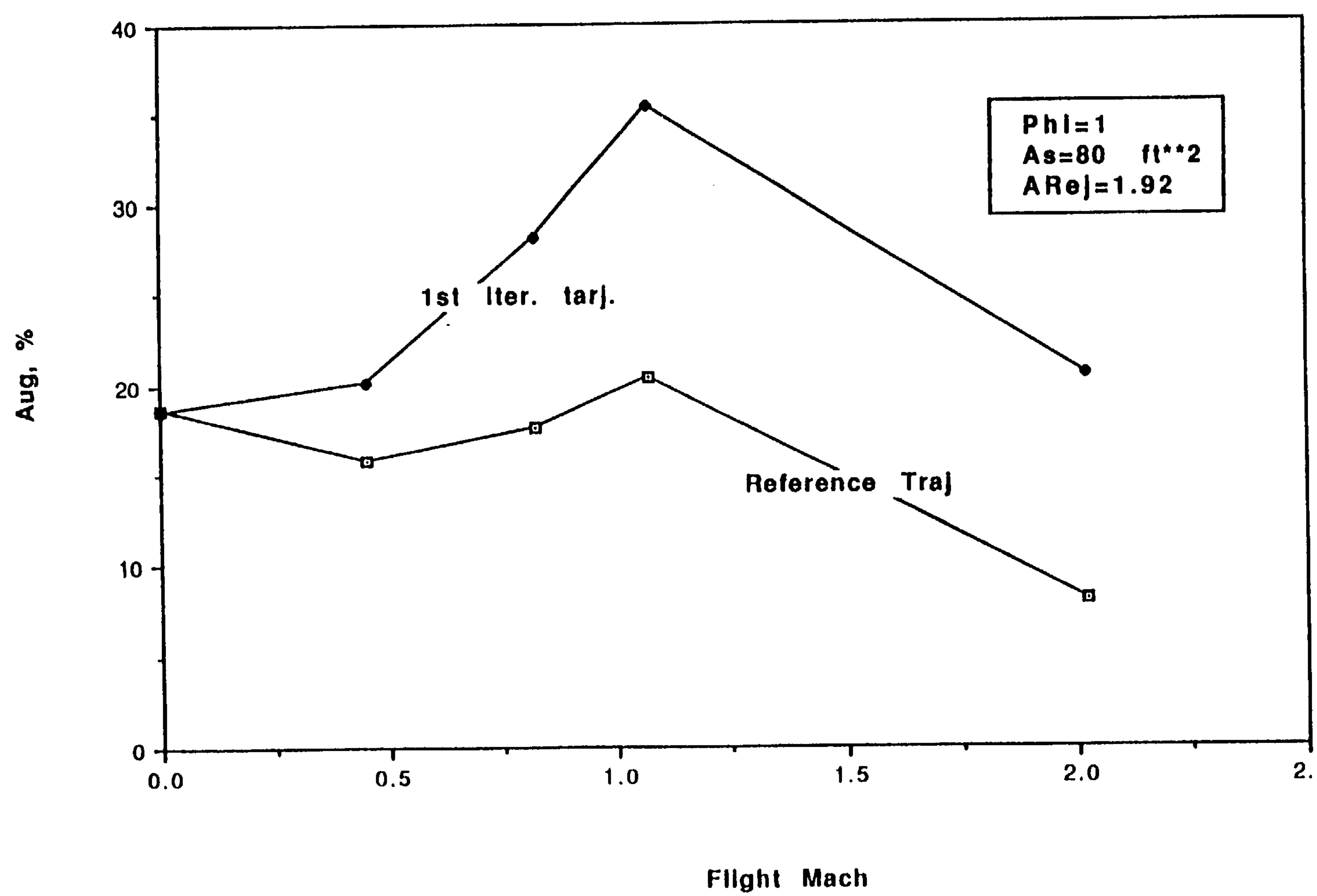
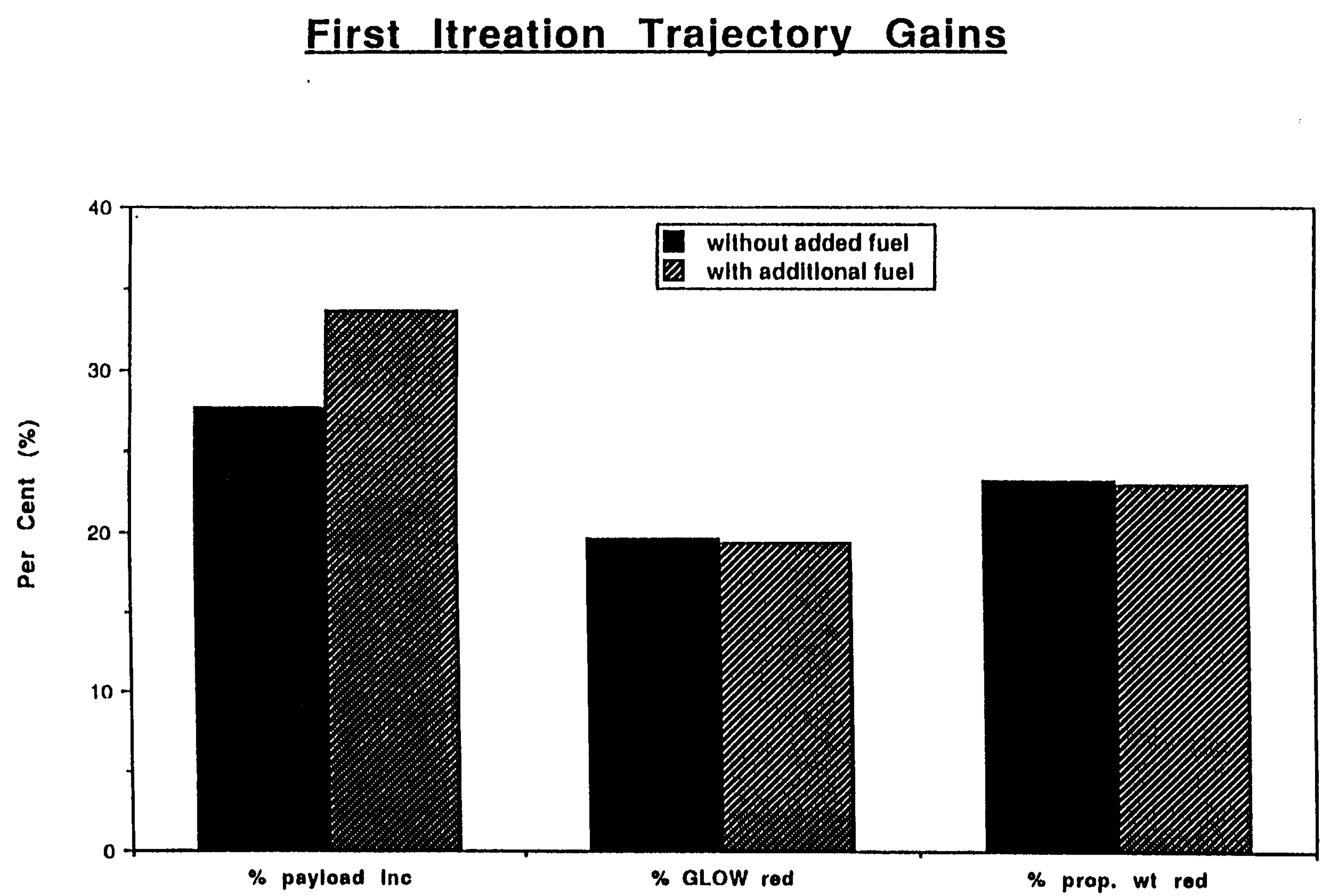


FIGURE 22.



NOTE: As=80 ft**2, ARej=1.63 & 1.92, Ref P/L=120 Klbs, Ref Traj=1st iter w/o fuel

FIGURE 23.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	Sep 92 - Aug 93	Final Data Books & Report Sep 92-Aug 93

4. TITLE AND SUBTITLE Operationally Efficient Propulsion Study; Vol VI, Space Transfer Propulsion Operationally Efficient Study/Task, Vol VII, LOI Design Features & Options; Vol VIII BPM Engine Start Dynamics, Vol IX Prelim Dev Plan for Int BPM * Con't

5. FUNDING NUMBERS
Contract NAS10-11568

6. AUTHOR(S) Vol VI Timothy J. Harmon; Vol VII James M. Ziese; Vol VIII Victoria R. Kemp; Vol IX Angelo G. DiBlasi; Vol X Shahram Farhang; Video Script George S. Wong & Glen S. Waldrop; Edited by Donnie Trent; ** Con't

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
Rockwetdyne Division
Rockwell International Corporation
6633 Canoga Avenue
P.O. Box 7922
Canoga Park, CA 91309-7922

8. PERFORMING ORGANIZATION REPORT NUMBER
RI/RD 90-146-6
RI/RD 90-149-7
RI/RD 90-149-8
RI/RD 90-149-9 & 10

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
NASA
Kennedy Space Center, FL 32899

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
N/A

11. SUPPLEMENTARY NOTES
These data books and final report represent the completion of the OEPS Study Contract and should be considered with original 5 volumes

12a. DISTRIBUTION/AVAILABILITY STATEMENT
No Restrictions/Unlimited

12b. DISTRIBUTION CODE

13. ABSTRACT (*Maximum 200 words*)
This document is a collection of technical data books on several tasks of study during a two year extension to the basic Operationally Efficient Propulsion System Study (OEPSS) conducted by Rocketdyne Division, Rockwell International for the NASA at Kennedy Space Center (KSC), Florida. This document is organized in the following data books: Volume VI, Space Transfer Propulsion Operationally Efficient Study Task of OEPSS; Volume VII, Launch Operations Index (LOI) Design Features and Options; Volume VIII, Integrated Booster Propulsion Module (BPM) Engine Start Dynamics; Volume IX, Preliminary Development Plan for an Integrated Booster Propulsion Module; Volume X, Air Augmented Rocket Afterburning; and an OEPSS Final Briefing/Report along with an OEPSS Video Script with Video.

* Block 4 Con't: Vol X, Air Augmented Rocket Afterburning; OEPSS Final Briefing/Report; OEPSS Video Script with Video
** Block 6 Con't: Final Briefing/Report, Glen S. Waldrop & Timothy J. Harmon; All the aboved Managed By R.P.Pauckert/G.S.Waldrop of Rocketdyne & Russel Rhodes of NASA

14. SUBJECT TERMS
See Org. RDP for these terms, Ref Contract NAS10-11568

15. NUMBER OF PAGES

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	Unlimited