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CONFIGURATION DEVELOPMENT STUDY OF THE X-24C HYPERSONIC RESEARCH AIRPLANE - PHASE II

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16. Abstract

The X-24C Hypersonic Research Vehicle, configured with a Lockalloy heat-sink structure, a launch mass limit of 31.75 Mg and powered by an LR-105 Rocket Engine plus 12 LR-101 Sustainer Engines, has been found to be the more cost effective of the candidate configurations. In addition, the configuration provides the maximum "off design" growth potential capability and subsequently, has been selected as the candidate configuration to be subjected to the design refinement study in the remaining segment of the study.

Selection of this configuration was based on the analytical study conducted on the performance growth capabilities of the candidate configurations selected from the Phase I Study.

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FOREWORD

This analytical study report is submitted to the National Aeronautics and Space Administration in accordance with NASA Contract NAS 1-14222. The work reported herein was performed between April 1976 through May 1976 culminating in an oral presentation at NASA LRC on 27 May 1976. The study was performed by the Advanced Development Projects "Skunk Works" of the California Company, A Division of Lockheed Aircraft, under the supervision of Mr. H.G. Combs, Study Manager. Engineering graphics and supporting text were developed under the direction of Messrs. D.H. Campbell (Propulsion and Thermodynamics), M.D. Cassidy (Aerodynamics), C.D. Sumpter (Structures), E.B. Seitz (Weight), G.J. Kachel and R.P. James (Vehicle Design), J. Walters and consulting services of J. Love (Maintenance), and R.T. Passon (Cost). The Program Monitor for NASA was Mr. J.D. Watts.

This study was a co-operative effort between the contractor and NASA in which data and frequent consultation, as well as program direction were provided by NASA.

SUMMARY

Phase II analytical study was performed on the performance growth potential of the research vehicle configuration's that emerged from the Phase I Study. The results of this study permitted selection of the cost effective configuration combination to be subjected to refinement during the last phase of this study program.

Four vehicle configurations, consisting of two different structure concepts, in combination with two propulsion systems, were subjected to a systematic program involving development and evaluation of varied performance envelopes for launch mass of 25.85 Mg through 31.75 Mg (launch vehicle limit), with and without scramjets.

Analysis on the problem of field maintenance of the X-24C vehicle was expanded upon during this phase with particular emphasis placed on the real world results of the X-15-2 maintenance program.

Trade study results reached by this study include the recommendation to drop the Ablator TPS and RSI in favor of a Lockalloy heat-sink structure due to the advantages and off design potentials of the Lockalloy. Also, recommended are a 31.75 Mg launch mass and the LR-105 plus 12 LR-101 engine/sustainer combination for the propulsion system.

The concluding phase of the study will include a conceptual aerodynamic, structural and vehicle refinement of the X-24C configured around the Phase II Study recommendations.

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INTRODUCTION

Trade studies conducted by a joint NASA/USAF ad hoc committee in 1974 established a number of configuration alternatives regarding propulsion, structures and thermal protection which a proposed research vehicle could accommodate.

To narrow these configuration options created by the configuration alternates NASA authorized, in November 1975, a study contract to determine the cost effectiveness of these configuration alternates. Thereby, NASA could select the most advantageous of the alternatives to aid in defining firm requirements for the proposed X-24C research vehicle.

The study contract consisted of a three-phased program. The first phase would provide cost and mass trade results of the alternate structures, thermal protection systems (TPS), and propulsion systems from which a single configuration would be recommended for evaluation in Phase II. Phase II would look into the performance growth of the recommended Phase I configuration along with the attendant cost associated with the increase performance. Phase III would study the refinement of the NASA/USAF X-24C aerodynamic configuration, used in Phase I and II, and conceptual design of the vehicle which evolves from the design trades and growth evaluation results from Study Phase I and II.

Conclusions reached in the Phase I study recommended the aluminum substructure with the elastomeric Ablator TPS and the heat-sink Lockalloy to be the most viable structural concepts - to be carried into the Phase II study due to their approximately equivalent acquisition cost. The propulsion system selected as the prime candidate was the Rocketdyne LR-105 rocket engine with twelve (12) Rocketdyne LR-101 Atlas vernier engines, due to the mass growth potential it provided. This would result in an advantage in determining an optimum launch mass for Phase III. The study also recommended the Thiokel LR-99 rocket engine with two (2) Thiokel LR-11 sustainer engines as a back-up to the prime selection. The Phase I Study was conducted by the Advanced Development Projects

"Skunk Works" of the California Company, A Division of Lockheed Aircraft Corporation. Trade studies and results are covered by NASA CR-145032 dated December 1976.

This report covers the Phase II study conducted on the performance growth of the research vehicle configurations which emerged from the Phase I Study and described in the Candidate Vehicle section herein. In addition to analysis supporting performance growth, of the candidate vehicles, analysis on vehicle maintenance developed in Phase I were expanded to reflect the impact due to performance growth. A detail review of the X-15 program was conducted to draw upon prior experience to support X-24C analogous. Likewise, review of prior testing on X-15 materials and more recent LRC aerothermal testing of TPS materials results were looked at with a very critical eye to support the results and recommendations that this study produced.

BASIS FOR DESIGN TRADES

The basic objective of the study effort was to determine, through a systematic trade study, the performance growth potential of the vehicle design concepts selected by NASA as an outgrowth of the results and recommendations from the Phase I Study. Expansion of the Phase I Study design and analysis was of sufficient depth to support the Phase II effort, but as was noted in the Phase I effort, entails further design effort before the design can support a manufacturing program. The concepts evaluated in this study meet the (1) aerodynamic requirements, (2) performance requirements, (3) research requirements, (4) operation requirements, (5) structural requirements, and (6) cost - assumptions established by NASA/USAF for the Phase I program and modified by the following input data set forth by NASA/USAF for this study:

Configuration - For purposes of the growth potential evaluation the vehicle configuration concepts shall be considered (1) clean without a scramjet test package, and (2) with cruise scramjets.

Performance - Performance desires are aimed at not only assessing the impact of using the rocket cruise fuel to boost the higher Mach numbers but also to determine the feasibility and cost of obtaining higher boost Mach numbers and corresponding longer cruise times at intermediate Mach numbers. Efforts to increase performance must be constrained by limits on fuel volume, thermal protection system limits - and vehicle costs.

Launch Mass - Launch mass of 25.85 Mg (present B-52 limit) and 31.75 Mg shall be assumed for this growth potential evaluation. The 31.75 Mg mass is a theoretical mass that may be attainable through modifications to the B-52 launch aircraft, which is being assessed separately (not a part of this study).

Phase II Results - The contractor shall determine the maximum boost Mach number, the vehicle dry mass, initial vehicle cost, and an assessment of TPS annual maintenance manpower and relative program risks of each concept studied in Phase II. The Contractor shall then, jointly with the Technical Monitor, select a concept for Phase III.

CANDIDATE VEHICLES

Considering that an increase in vehicle performance was affected by the selected Mach number, variation in dynamic pressure, variation in load factor, and type of mission a spectrum of configuration concepts emerged from the four potential concepts recommended in the Phase I Study. All Phase II conceptual alternatives were based on the propulsion and structural system combinations emerging from the Phase I Study, which included:

Propulsion Systems:

(1) Rocketdyne LR-105 engine with twelve (12) Rocketdyne LR-101 Atlas vernier engines, or

(2) Thiokel LR-99 rocket engine with two (2) Thiokel LR-11 sustainer engines.

Structural Systems:

(1) A heat-sink substructure compatible with a Lockalloy panel skin surface, or

(2) An aluminum skin and substructure covered over with an Ablator TPS.

TECHNICAL APPROACH

A systematic trade-off analysis on the growth potential of each of the candidate configuration concepts was conducted in sufficient depth to support the analysis, results and recommendations delineated herein. The trade-off analysis process followed the activities diagrammed in Figure 1 which include the following main tasks:

Task I - Developed mission profiles and maximum zero fuel mass for the 31.75 Mg launch mass vehicle, with and without scramjets. This then allowed a direct comparison with the 25.85 Mg launch mass vehicle developed in Phase I of the study.

Task II - Developed performance envelopes as a function of cruise time versus Mach number for the various critical parameters such as propulsion performance capability, thermal protection capability, zoom capability, varied dynamic pressure, varied load factor, drag reduction, etc. These performance envelopes were developed for launch masses of 25.85 Mg through 31.75 Mg, with and without scramjets.

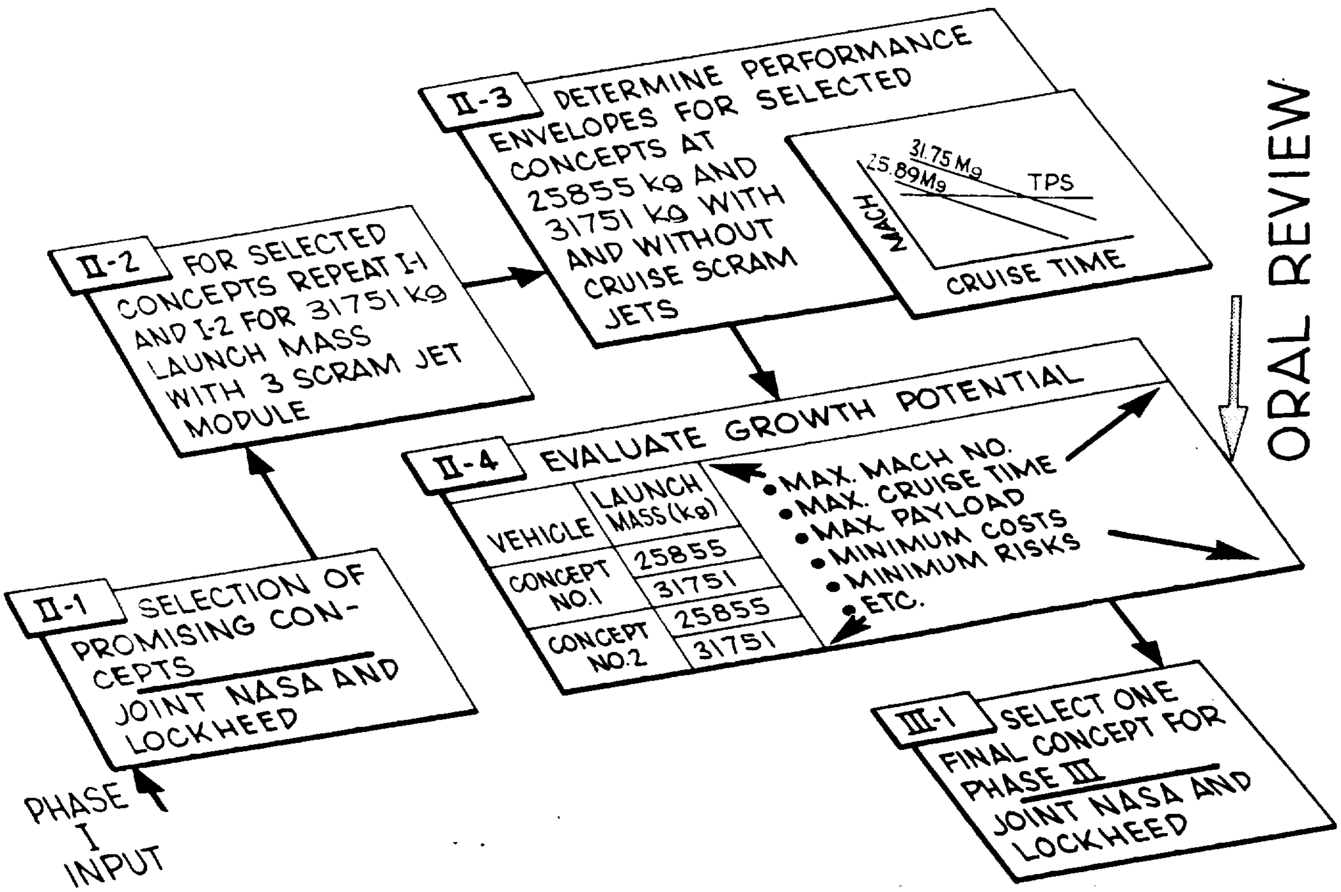


Figure 1 - Growth Potential Design Trades

Task III - Based on the performance envelopes developed in Task II a comparison table was developed to evaluate the growth potential of each of the candidate configurations.

Task IV - Refined the Phase I Maintenance Analysis by drawing upon the X-15 program and the Langley Research Center (LRC) TPS test programs to support X-24C analogous regarding vehicle systems maintenance and TPS maintenance.

Task V - Expanded the Phase I Costing Analysis to assist in evaluating the growth potential of the candidate configurations.

Task VI - Based on the data developed at this point, recommendations were made relative to the selection of a concept to be pursued in the final study phase.

MASS ANALYSIS

In support of the investigating analysis produced in this study, mass analysis on a number of potential vehicle concepts were conducted. Using as a Baseline the four concepts which emerged from the Phase I study, referred to in the Candidate Vehicle section of this report, and coupled with the parameters established for this study produced a spectrum of 384 vehicle concepts. The Phase II parameters included (1) two vehicle launch masses of 25.85 Mg and 31.75 Mg, (2) three scramjet module arrangements consisting of zero, 3 and 8 units, (3) four Mach numbers of 5, 6, 7 and 7-1/2, and (4) four cruise range periods of zero, 40, 80 and 120 seconds. Other parameters, such as, (1) cruising with rockets vs cruising with scramjets, (2) higher Mach numbers, (3) variation in dynamic pressure, and (4) variation in load factor were included increasing the spectrum of vehicle concepts analyzed to well over 500.

Starting with the Baseline concepts each was sized to fit a specific mission emerging from the combination of parameters. Sizing took the form of lengthening of the baseline fuselage body, as illustrated in Figure 2, according to the propellant volume necessary to complate a specified mission. These results were utilized in part in the various study analysis described herein.

Table 1 lists a typical sampling of vehicle concept mass generated using mass analysis described above and illustrated as follows:

Method of Analysis

Step 1 - Selecting one of the Baseline vehicles provides the Operating Mass Empty (O.M.E.) and propellant mass required to do a mission consisting of 40 seconds cruise at Mach 6 with three (3) scramjet module units mounted, but inoperative, at a launch mass of 25.85 Mg.

Step 2 - List the propellant mass required to do the specified missions. These numbers were derived from the aerodynamic analysis which defined the

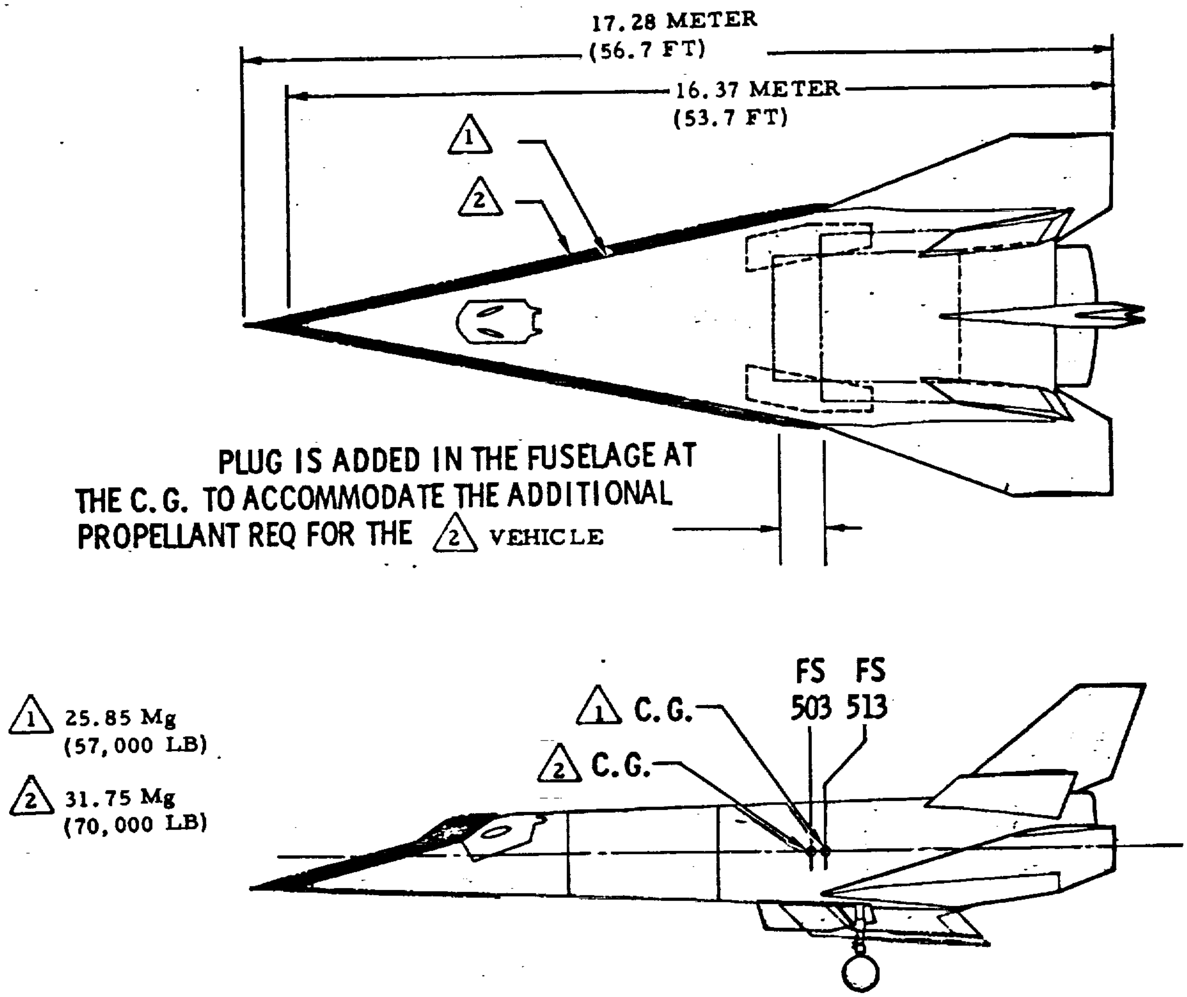


Figure 2 - Baseline Vehicle Sizing

Concept	Cruise Mach	Boost Fuel kg x 10^{-3} (lb)	Cruise Time (sec)	Rocket Cruise Fuel kg (lb)	Total Mission Fuel kg x 10^{-3} (lb)	TPS Penalty kg (lb)	Fuel O.M.E. Penalty kg (lb)	Operate Mass Empty kg x 10^{-3} (lb)	Scramjets No./Mass kg (lb)	Additional Payload kg x 10^{-3} (lb)	Scramjet Inlet Area m^2 (ft^2)
1 △1	6	13.77 (30,357)	40	1329 (2,930)	15.1 (33,287)	0	0	8.91 (19,650)	1225 3/(2,700)	0.62 (1,363)	0.5 (5.4)
2 △2	5	15.52 (34,223)	0	0	15.52 (34,223)	-550	85 (187)	8.75 (19,287)	2517 8/(5,550)	4.96 (10,940)	1.6 (17.3)
3 △2	6	17.39 (38,345)	40	1468 (3,236)	18.86 (41,581)	(300)	753 (1,659)	9.8 (21,609)	2630 8/(5,800)	0.46 (1,010)	
4 △2	7	19.0 (41,933)	0	0	19.0 (41,933)	(-40)	784 (1,729)	9.68 (21,339)	2563 8/(5,650)	0.49 (1,078)	
5 △2	6	17.39 (38,345)	40	△4	17.39 (38,345)	(300)	459 (1,012)	9.51 (20,962)	2734 8/(6,028)	2.1 (4,665)	
6 △2	6	17.39 (38,345)	80	2936 (6,472)	20.3 (44,817)	(1,430)	1046 (2,306)	10.6 (23,386)	2722 8/(6,000)	-1.91 (-4,203)	
7 △2	7.5	19.75 (43,558)	0	0	19.75 (43,558)	(420)	932 (2,054)	10.0 (22,124)	2574 8/(5,675)	-0.62 (-1,357)	1.6 (17.3)
8 △3	6	14.69 (32,395)	40	1139 (2,511)	15.8 (34,906)	0	0	9.0 (19,918)	1225 3/(2,700)	-0.24 (-524)	0.5 (5.4)
9 △3	5	13.35 (29,441)	0	0	13.35 (29,441)	(-85)	-496 (-1,093)	8.5 (18,740)	2517 8/(5,550)	1.48 (3,269)	1.6 (17.3)
10 △3	6	14.89 (32,827)	0	0	14.89 (32,827)	(-85)	-189 (-416)	8.8 (19,417)	2540 8/(5,600)	-0.38 (-844)	
11 △3	6	14.89 (32,827)	40	1219 (2,688)	16.1 (35,515)	0	55 (122)	9.09 (20,040)	2630 8/(5,800)	-1.97 (-4,355)	
12 △3	7	16.22 (35,766)	0	0	16.2 (35,766)	(35)	78 (172)	9.13 (20,125)	2567 8/(5,660)	-2.06 (-4,541)	
13 △3	6	14.89 (32,827)	40	△4	14.89 (32,827)	0	-21 (-46)	8.8 (19,502)	2734 8/(6,028)	-0.62 (-1,357)	1.6 (17.3)

△1 Baseline vehicle 25.85 Mg (57,000 lb) launch mass, Lockalloy structure, LR-105 engine

△2 Identical to △1 except 31.75 Mg (70,000 lb)

△3 Baseline vehicle 25.85 Mg (57,000 lb) launch mass Aluminum/Ablator structure, LR-99 engine

△4 Cruise on scramjets

Table 1 - Vehicle Masses

propellant required to reach a desired speed and propellant flow per second of cruise time for the various conditions of launch mass, Mach numbers, cruise time and scramjet arrangement.

Step 3 - List the Lockalloy/TPS mass penalties derived from the thermodynamic analysis which established the incremental mass increase or reduction from the Baseline concept for changes in Mach number and cruise time. Thermo-structural considerations took into account temperature gradients, minimum thickness (Ablator) requirements, and number of scramjet modules.

Step 4 - Develop the Operating Mass Empty (O. M. E.) penalties by sizing the fuel volume requirement from Step 2. Thus the vehicle fuselage was lengthened or shortened by the amount dictated by the fuel tank size. This established a delta mass consisting of fuselage structure, tank shell, and system (controls/electrical/ etc.) to accommodate the change in fuel quantity. After several detail mass calculations using this procedure it was established that a 5 to 1 ratio of O. M. E. penalty to fuel delta mass from the Baseline could be used as quick and relatively accurate method of computation.

Step 5 - By adding the Lockalloy/TPS mass penalty (Step 3) and the O. M. E. penalty to the Baseline vehicle O. M. E. (Step 1) a new O. M. E. was derived for each vehicle that perform a single designated mission.

Step 6 - Adding the new O. M. E. to the mass of the propellant (total mission fuel) Step 2 and subtracting this sum from the starting launch mass gave a difference identified as "Additional Payload." For missions using scramjets the mass of scramjets with its cruise fuel is summed up with the O. M. E. and propellant mass before subtracting from the launch mass in order to arrive at the "Additional Payload" which was then used as an indicator. If the number came out positive, it showed that it is possible to build the vehicle to do the specified mission and how much mass was left over (margin between total mass required for the mission and the starting launch mass). If the number came out negative it indicated that it was impossible to build the vehicle to do the mission within the constraint of the launch mass. For example, starting with one of the Baseline vehicles, concept 1, Table 1, we have a configuration with a launch mass of 25.85 Mg, a Lockalloy structure, a propulsion system consisting of an LR-105 engine and 12 LR-101 vernier engines, 15.10 Mg of propellant and an O.M.E. of 8.91 Mg, capable of 40 seconds cruise at Mach 6. This vehicle has a 618 kg "Additional Payload" margin. For a 31.75 Mg launch mass vehicle (concept 5, Table 1) similarily configured (except with 8 in lieu of 3 scramjet modules) to do the same mission requires 17.39 Mg of propellant. This is 2.29 Mg more than the Baseline vehicle, thereby requiring a larger (longer) fuel tank and vehicle fuselage. Using the 5 to 1 mass ratio adds 459 kg of vehicle/tank mass to the O.M.E. From the Lockalloy TPS increment mass curve, Figure 48, the penalty is found to be 136 kg. Combining the Baseline O.M.E. (8.9 Mg), the added fuselage/tank structure (459 kg) and 136 kg of TPS penalty produces a new O.M.E. of 9.51 Mg. Adding the new O.M.E. mass with the required propellant mass of 17.39 Mg and the mass of the scramjet package with its cruise and cooling fuel (2.73 Mg) results in a total launch mass of 29.64 Mg. This is 2.11 Mg less than the 31.75 Mg launch limit. Conclusion, therefore, being that this concept can be built, can do the mission and has 2.11 Mg to spare for "Additional Payload." Another example, concept 13, Table 1, with the LR-99 engine with the Ablator at a launch mass of 25.85 Mg to cruise at M = 6 for 40 seconds on eight scramjets requires 14.9 Mg of propellant. This is 943 kg less propellant than the base airplane, concept 8, Table 1, and would reduce its O.M.E. of 9.03 Mg by 189 kg to 8.85 Mg. Adding this O.M.E. of 8.85 Mg to propellant required of 14.89 Mg to scramjets 2.73 Mg totals 26.47 Mg which gives the "Additional Payload" mass of minus 616 kg, indicating that this vehicle could not be built to do the mission within the 25.85 launch mass constraint.

B-52 AIRCRAFT CONSTRAINTS

To support the analysis for determining the performance growth of the Phase I Baseline X-24C it was necessary to evaluate the physical and operational constraints presented by the existing B-52 launch aircraft and the effect of the constraints on the growth potential of the X-24C.

B-52 definition was provided by the North American Aviation drawing 2581-900901 titled: General Arrangement B-52/X-15A-2. This document defines the present B-52 provisions for the X-15A-2 aircraft, but does not take into account the results of a separate study by Boeing Aircraft to upgrade these provisions. The data developed for this study regarding the B-52 interface was coordinated with representatives of the Boeing Company.

Figures 3 and 4 provide an overview of the major elements of clearance investigated. They are (1) physical clearance to the B-52 fuselage, wing lower surface, and wing cutout for the X-24C vertical fin, (2) ground clearance, (3) B-52 engine jet wake impingement on the X-24C, (4) X-24C ejection seat path clearance, (5) B-52 pylon station load limits, and (6) X-24C center of gravity location limits relative to the B-52 pylon attachment. Figure 5 delineates the physical clearance envelope developed of the B-52 based on the N. A. A. B-52 drawing with the X-24C shape superimposed in Figure 6. The ground lines were established by using the worst possible vehicle landing conditions, i. e., forward gear static and aft gear fully compressed for fore and aft clearances; and one main gear fully compressed and all others static for side-to-side ground clearance. The X-24C c. g. location, relative to the B-52 was coordinated with Boeing engineering. The 0. 3 meter gap between the B-52 wing and X-24C top surface is based on the assumption that a new B-52 pylon will be provided to support the heavier X-24C and physical envelope differences over the X-15A-2. The angle of incidence shown in Figure 6 results in the minimum aerodynamic loading on the B-52, however, by decreasing the angle of incidence an enhancement in ground clearances will be achieved along with an increase in B-52 aerodynamic loading. Jet wake impingement depicted in Figures 4, 5 and 6 were based on estimates of the jet wake plume of the B-52 engines. The seat ejection path clearance, to the B-52 aircraft, were based on time study analyses of the SR-71 seat capability.

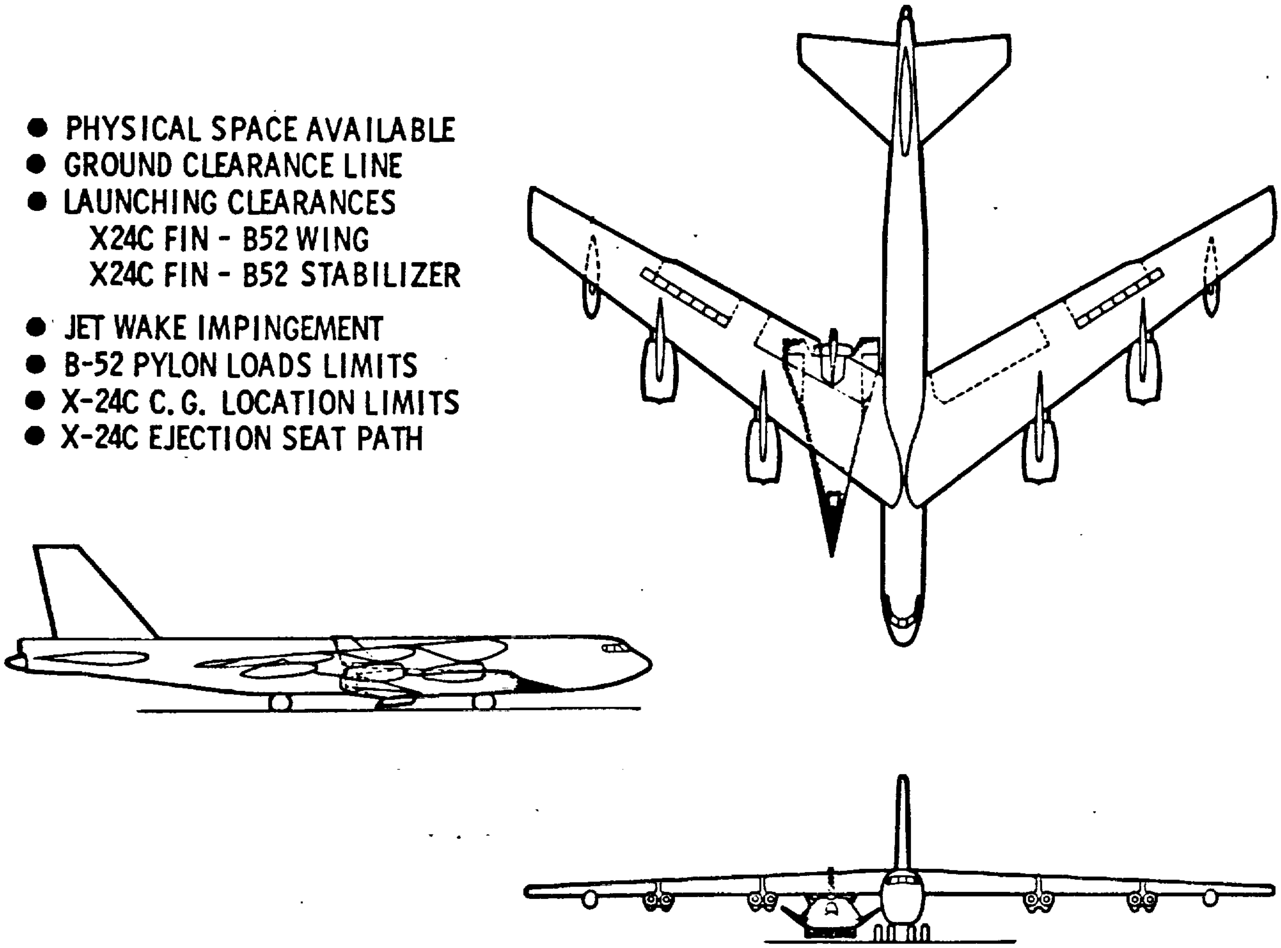


Figure 3 - B-52 Constraints

The above physical constraints, other than resulting in an extremely tight envelope present no major impact on the X-24C envelope resulting from this study.

Of note, results of the independent Boeing B-52 study must be looked at in light of the results of this analysis particularily if the launch station is moved outboard. A shift of this sort will provide more clearance with the B-52 fuselage but will cause more of X-24C to be impinged on by the jet wake and result in less ground clearance in the one wing down, Figure 6, condition.

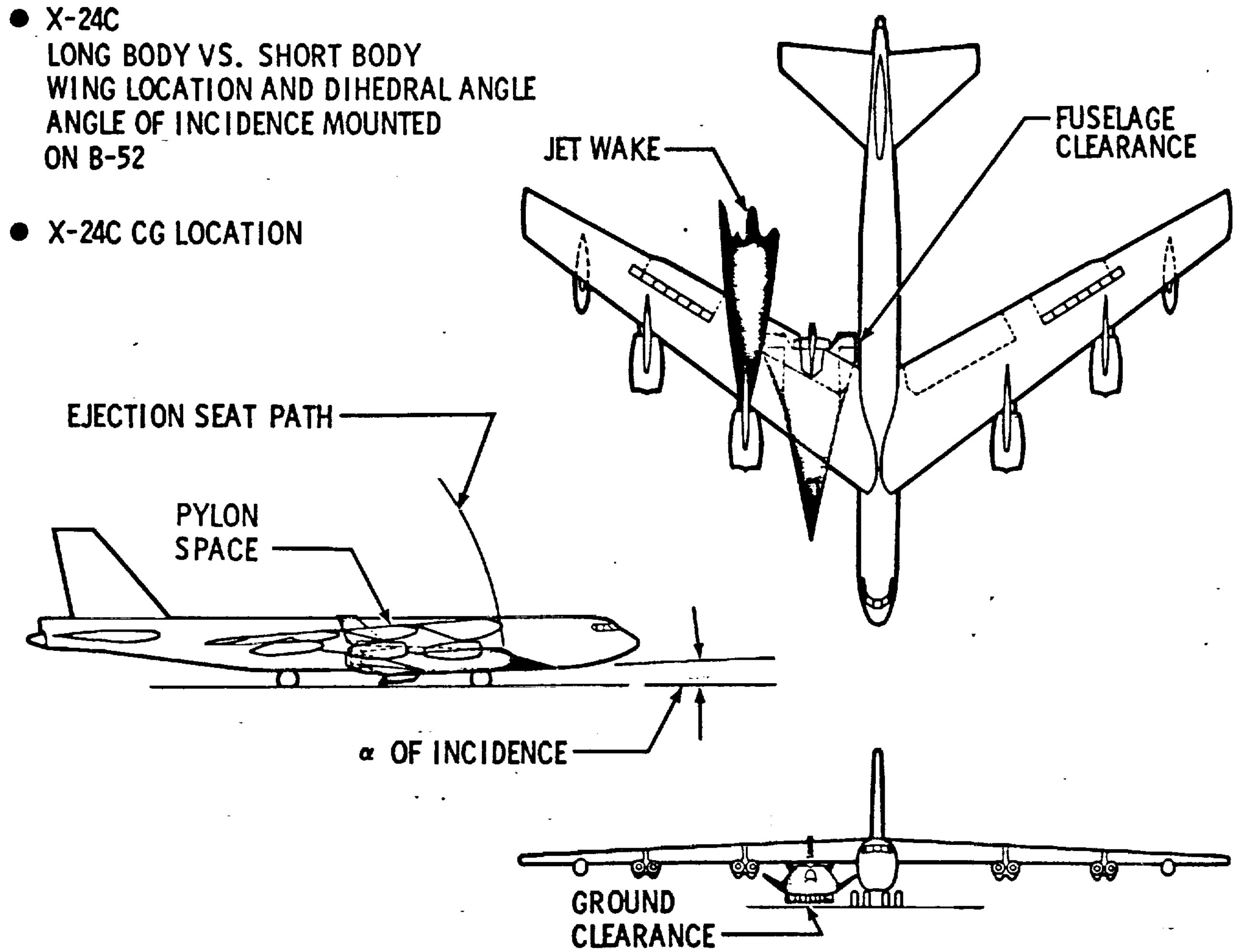


Figure 4 - B-52 Clearances and Constraints

MAXIMUM MACH NUMBER ATTAINABLE

Analysis relating to the maximum Mach number that the X-24C can obtain was conducted for two selected design conditions. The design conditions consisted of 40 seconds of scramjet cruise and boost to maximum Mach number. Variable parameters considered were launch mass, thermal protection system (TPS), and boost engine. Fixed parameters consisted of a maximum maneuver load factor of +2.0 g's, a dynamic pressure at end of boost of a 47.9 Pa and the candidate vehicle configuration.

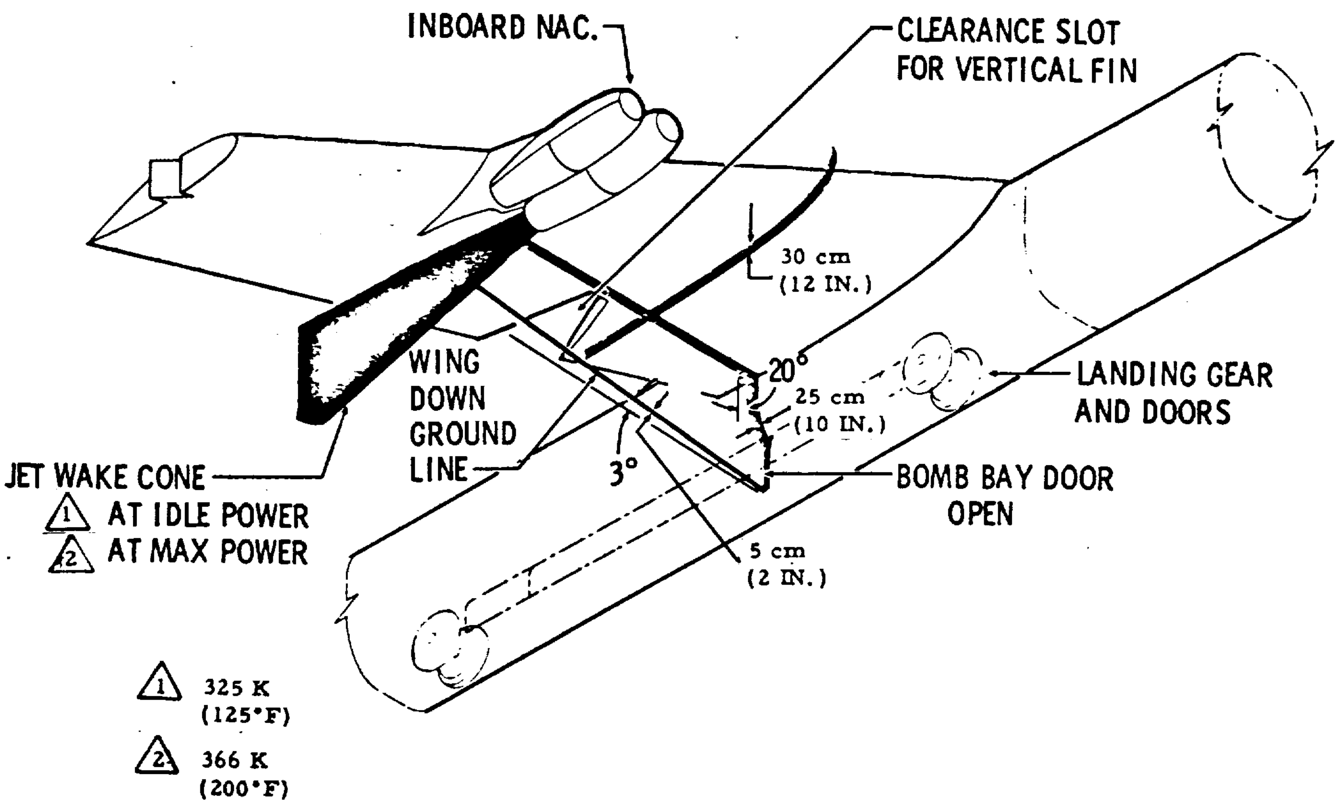


Figure 5 - B-52 Constraints Physical Space Available

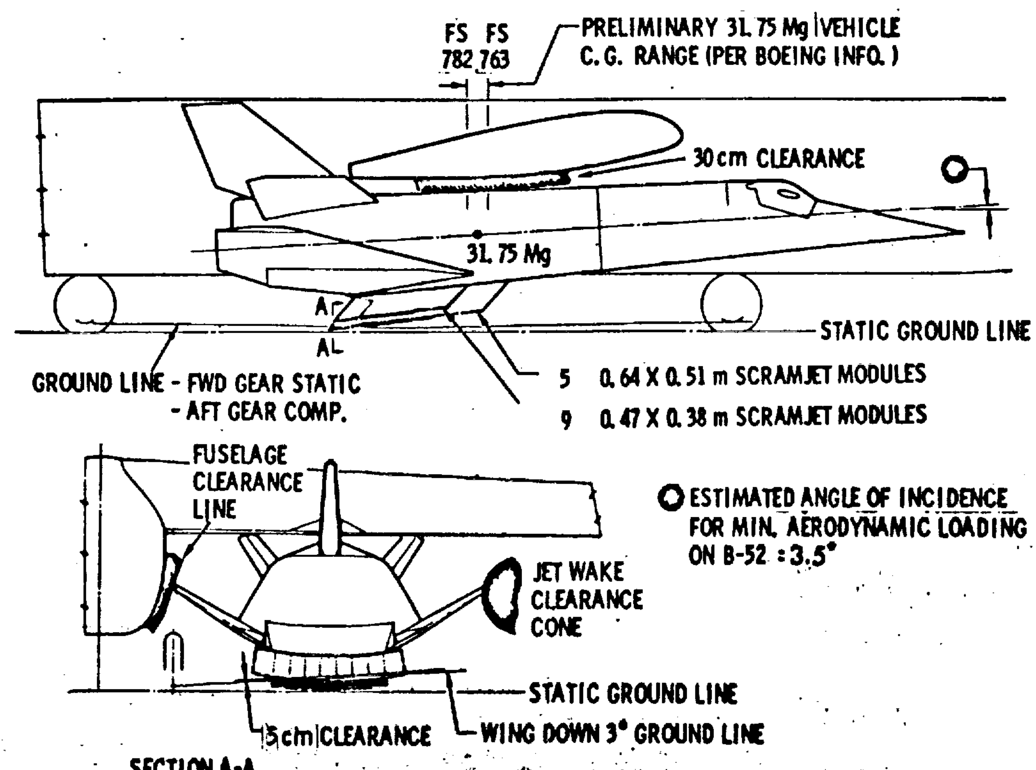


Figure 6 - X-24C/B-52 Interface

Figures 7 through 14 graphically represent the results of this analysis with the maximum Mach number for the design conditions investigated highlighted in bold type. The lesser of the two Mach numbers represents a vehicle capable of 40 seconds scramjet cruise. Its flight profile consisting of a rocket boost to the Mach number indicated, combined with 40 seconds of cruise on scramjet. Specifically, the fuel and TPS have been tailored within the design to this mission. It is noted that to perform this mission no additional payload capability is available. Off design capabilities cannot be realized from these figures, but are contained within the section "CAPABILITY VERSUS COST SUMMARY" herein.

The larger Mach number given in the figures represents the design of a vehicle specifically to perform a mission of boost to the maximum Mach numbers possible. This vehicle includes a 454 kg of payload on-board and is not capable of cruise at the achieved Mach number. However this does not preclude cruise at lower Mach numbers.

The method used to determine the limits depicted in Figure 7 through 14 was to construct curves of vehicle mass segments as a function of Mach number. Each plot was assembled such that the intersection of the boost fuel segment (total energy) with the operating mass segment established the Mach limit for an initial launch mass and design condition. The vertical summation of the mass segments at a particular Mach number represents a specific design point vehicle.

Interpretation of a particular plot, depicted by Figure 7 through 14 as to the magnitude of a mass segment can be made. The lower portion represents the absolute value of the kilograms of fuel energy required. The total energy or rocket boost fuel has been divided into two absolute segments of aerodynamic drag energy and inertia energy to illustrate the relative magnitude of each contribution towards the total. The upper portion of the figures contains the fixed mass items such as vehicle structure, tankage, TPS, etc. and is measured from the launch mass down. Therefore, to obtain the kilograms of operating mass desired, the representative 'minimum operating mass empty' value is subtracted from the launch mass.

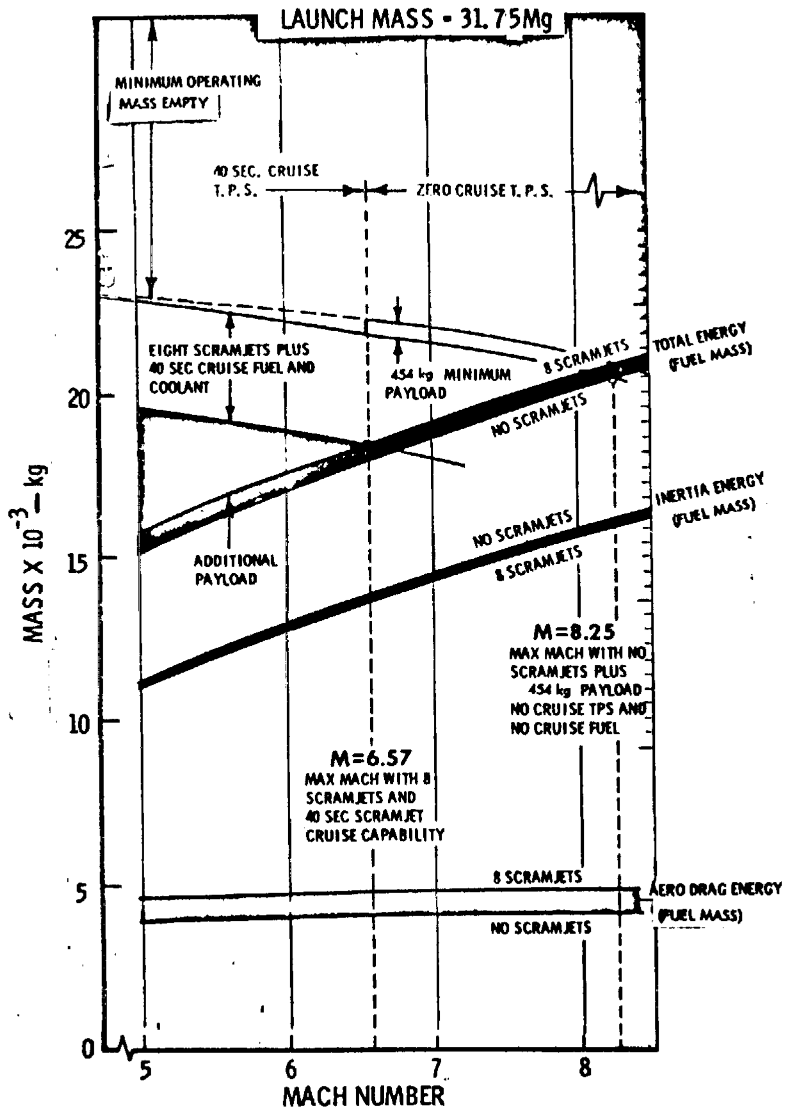


Figure 7 - Maximum Mach Number Attainable LR-105, 31.75 Mg - Lockalloy

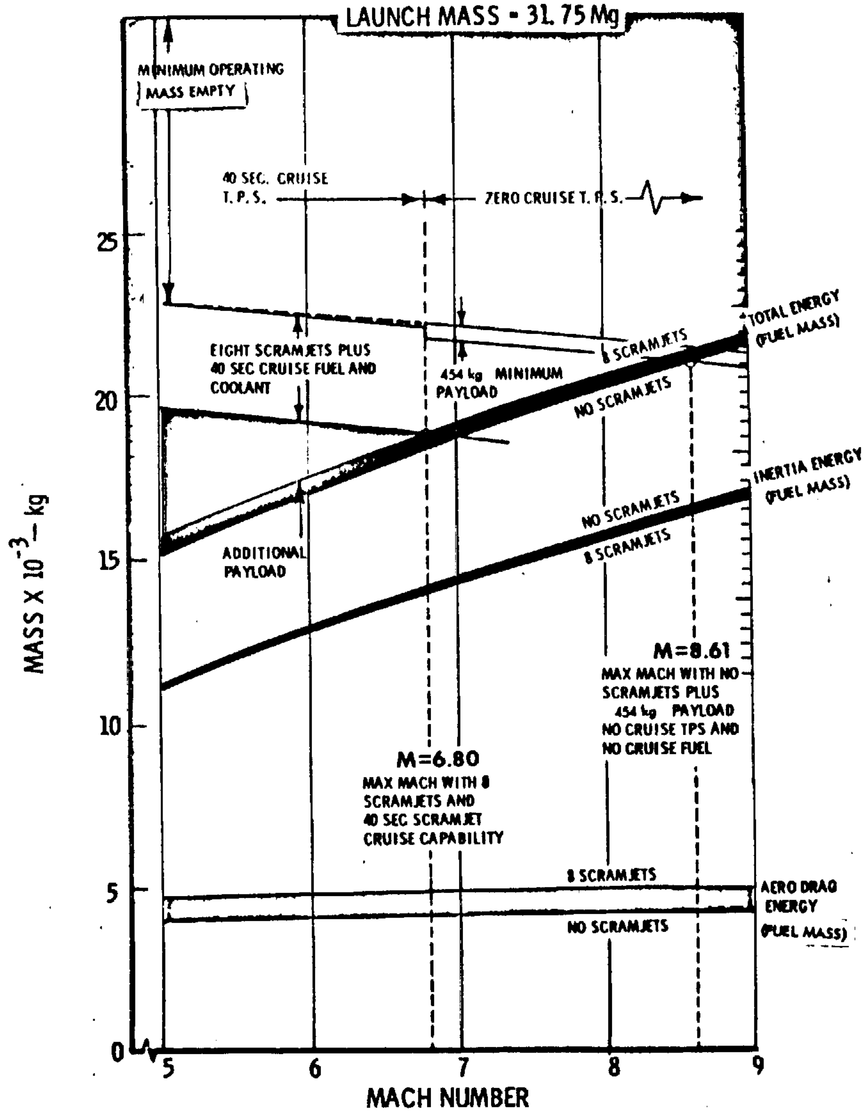


Figure 8 - Maximum Mach Number Attainable LR-105, 31.75 Mg - Ablator

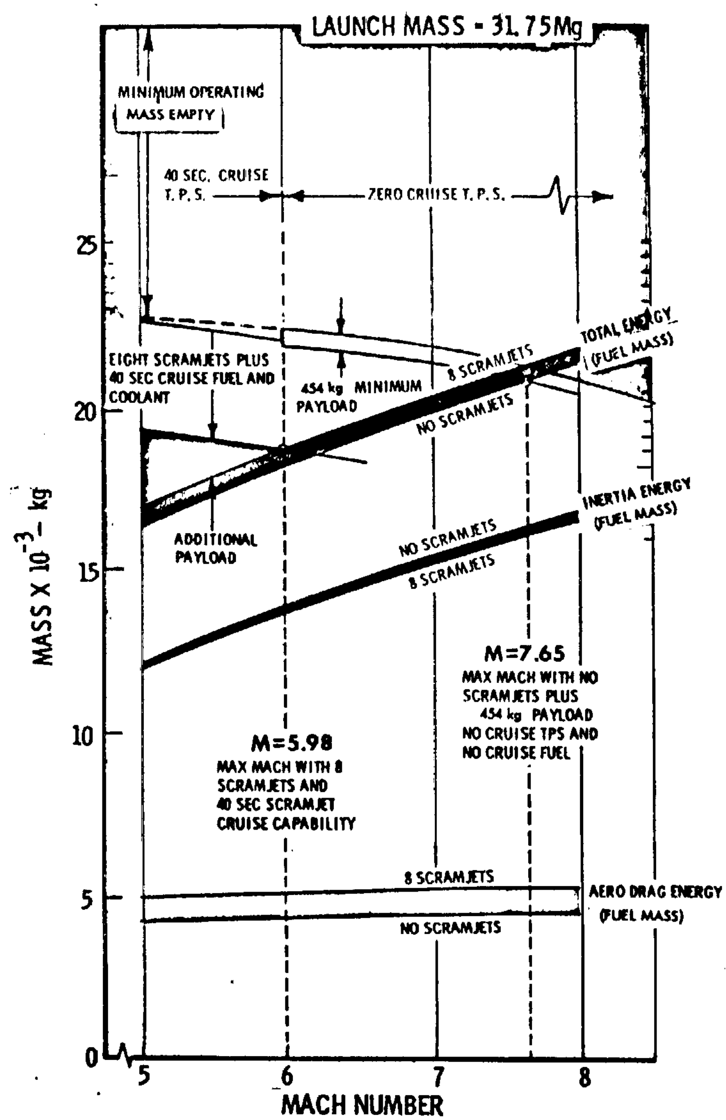


Figure 9 - Maximum Mach Number Attainable LR-99, 31.75 Mg - Lockalloy

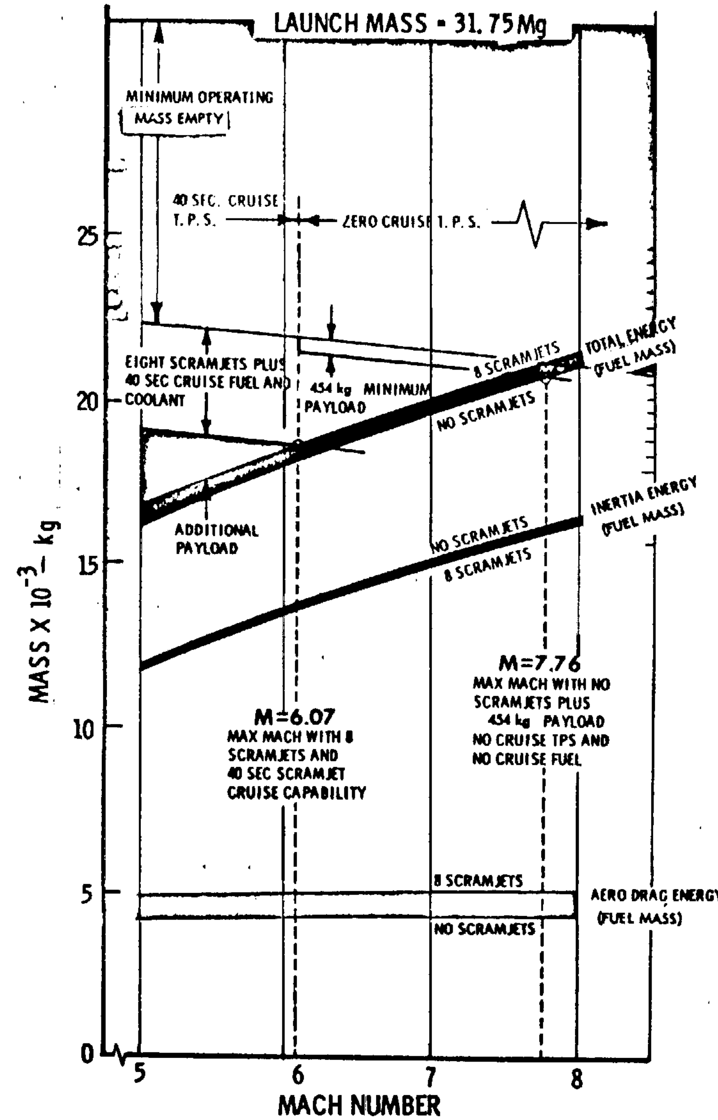


Figure 10 - Maximum Mach Number Attainable LR-99, 31.75 Mg - Ablator

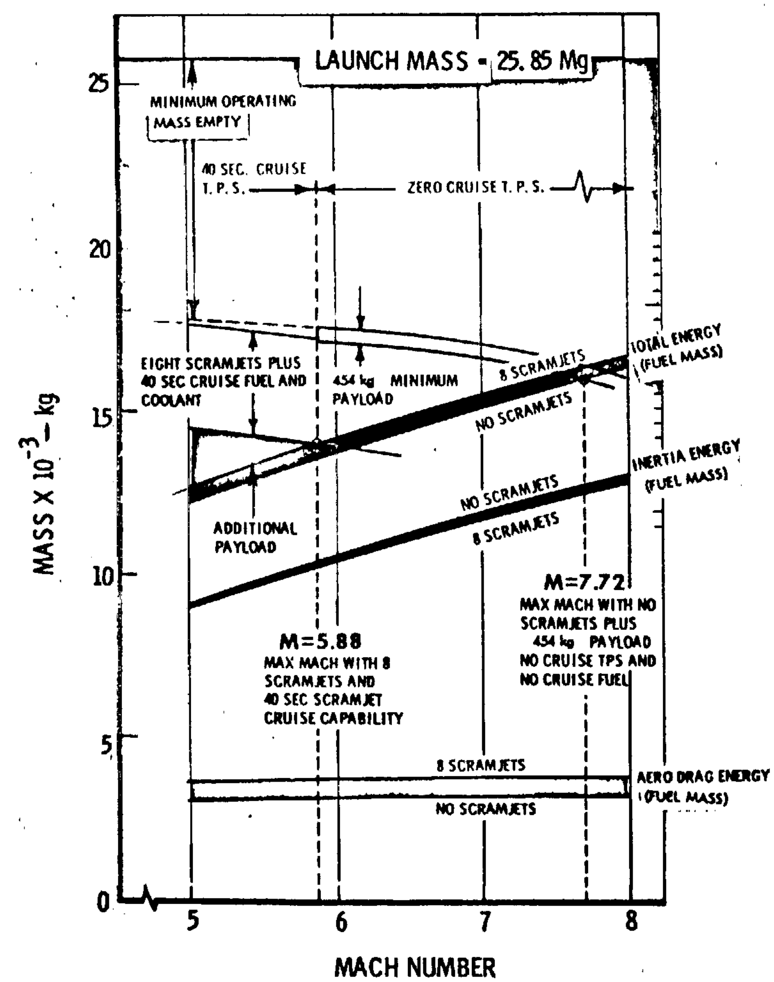


Figure 11 - Maximum Mach Number Attainable LR-105, 25.85 Mg - Lockalloy

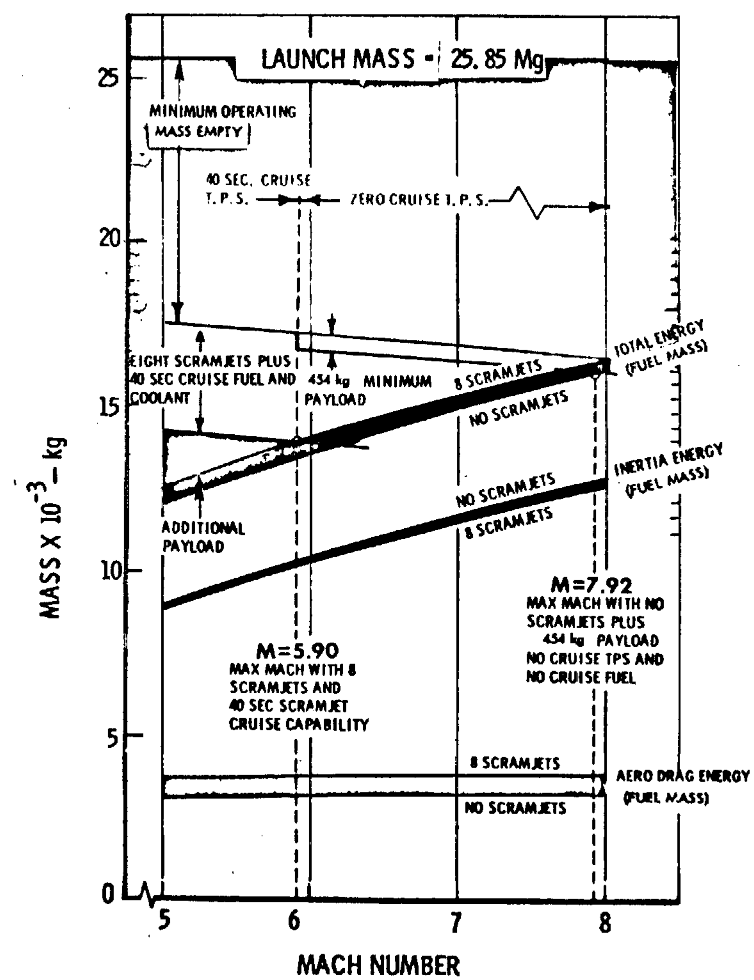


Figure 12 - Maximum Mach Number Attainable LR-105, 25.85 Mg - Ablator

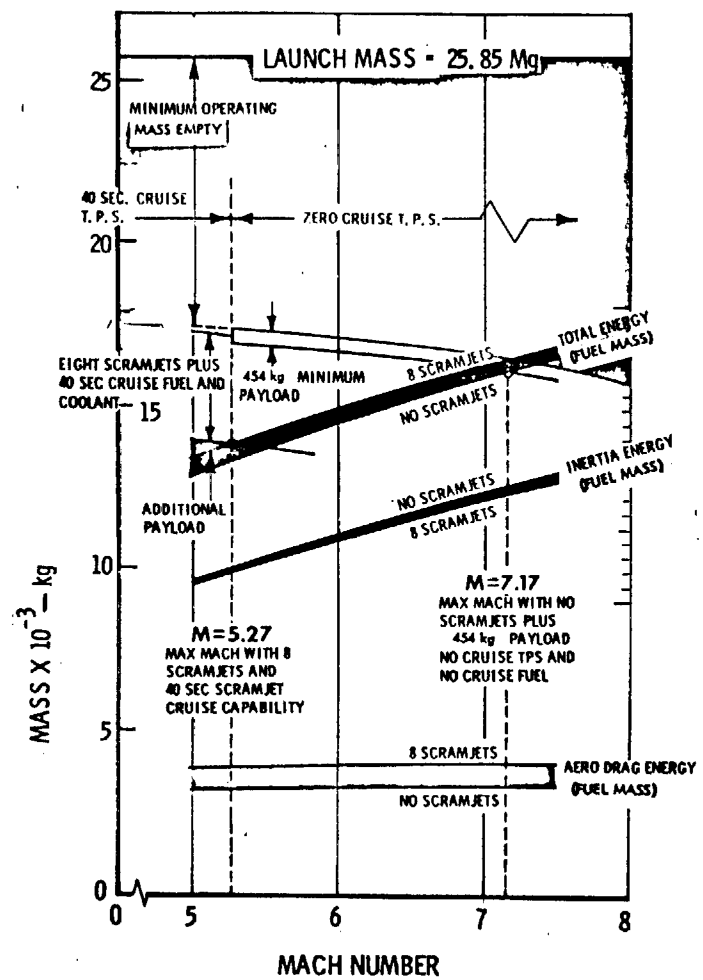


Figure 13 - Maximum Mach Number Attainable
LR-99, 25.85 Mg - Lockalloy

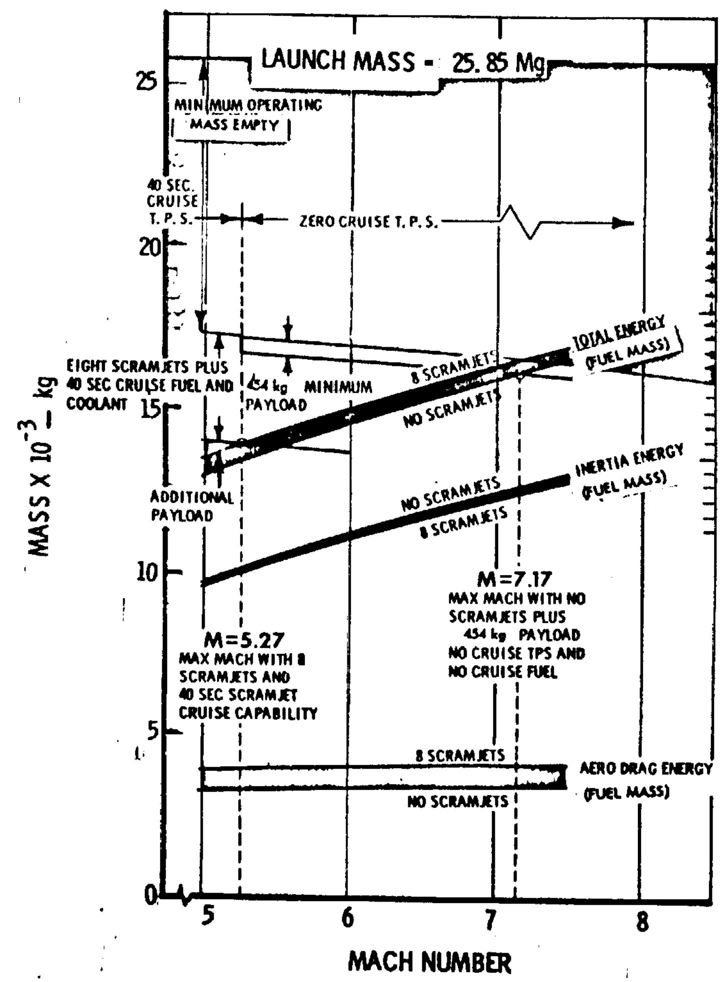


Figure 14 - Maximum Mach Number Attainable
LR-99, 25.85 Mg - Ablator

The analytical procedure consisted of the computer analysis program developed in Phase I of this study and used to map the mission profile from launch to the end of boost, for several Mach numbers. Details as to the mechanics of this computer program are covered in the "MISSION PROFILE" section of the Phase I report, Reference 1. Boost phase profile constraints were those noted in the lead paragraph herein. The computer analysis established the required fuel mass necessary to complete the boost phase. The fuel usage quantity is independent of the specific mission, but is dependent on the vehicle configuration (specifically the scramjet drag increment), engine Isp and launch mass. The computer analysis also allowed the separation of the contributions of each of the aerodynamic and inertia fuel usage. This became valuable when considering the effects of aerodynamic drag reduction.

Due to lower net energy level of the vehicle with scramjets at the end of boost, less energy is required to overcome the inertia effects. This is apparent in review of Figures 7 through 14, in that a larger inertia fuel quantity is required for the vehicle without scramjets.

The "minimum operating mass empty" represents all of the vehicles fixed mass items other than payload. The difference in the operating mass between cruise and non-cruise vehicles is due to the TPS and structural differences. "MASS ANALYSIS" section herein provides details on these differences.

Results of the analysis, Figures 7 through 14, indicates, regardless of the launch mass, the LR-105 with 12 LR-101 engine vehicle with Ablator TPS, Figures 8 and 12, can obtain the higher Mach number for scramjet cruise. Approximately 1/2 Mach number difference exists between scramjet cruise vehicles powered by either the LR-105 with 12 LR-101's or LR-99 with 2 LR-11's. The difference in Mach number caused by the Ablator and Lockalloy TPS is relatively small and is approximately 1/8 Mach number difference in the higher launch masses.

The same observation can be made with regard to vehicles designed to obtain maximum Mach number without scramjets. The difference in Mach number caused by the TPS differences is approximately 1/3 Mach number in the higher launch masses. In addition, the 25.85 Mg vehicle with the LR-105 and 12 LR-101 engines has a performance equal to the 31.75 Mg, LR-99 engine vehicles.

ROCKET CRUISE CAPABILITIES

Figures 15 through 30 represent the results of the analysis conducted to determine the "Additional Payload" capacity vs total design cruise time capability of the X-24C. "Additional Payload" has been defined as that amount of payload that can be carried on board the vehicle in addition to the fixed load (454 kg of instrumentation equipment) or cruise scramjets. Total design cruise time has been defined as that cruise time for which the vehicle has been designed and does not include additional scramjet cruise. Sustainer rocket thrust is used to power the vehicle through the cruise segment and TPS mass do not provide for a scramjet cruise phase. The figures graphically depict the design performance possibilities of the various engine, TPS and launch mass combinations considered within this analysis.

Development of Figures 15 through 30 was similar to those developed for the "Maximum Mach Number Attainable" herein, except the abscissa is total design cruise time. The ordinate represents the summation of the various components required to complete the defined mission, in terms of kilograms of mass, for a specific cruise time and Mach number. "Total energy expended" represents the total mass of fuel required for boost by the primary engines plus the mass of fuel consumed to cruise by the sustainer rocket engines. This value can be found in its absolute form in the lower portion of each figure. The upper portion represents the fixed mass items or the "minimum operating mass empty" plus a 454 kilograms of test instrumentation or cruise scramjets. The mass segment bounded by these two mission requirements is the "additional payload." As the design cruise time is increased, the additional payload converges toward zero where the maximum possible design cruise time is established.

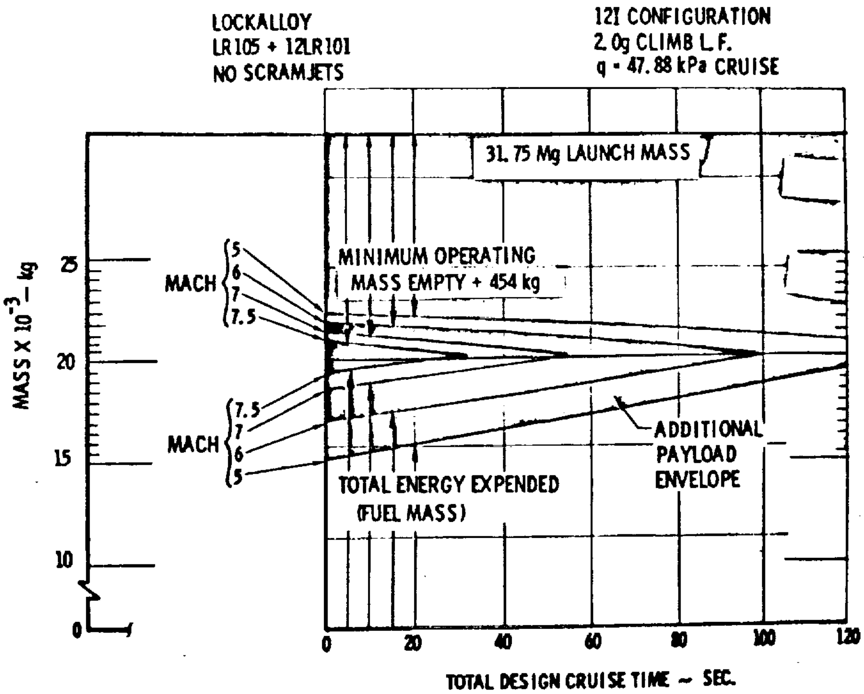


Figure 15 - Rocket Cruise Capabilities - LR-105, 31.75 Mg, No Scramjets - Lockalloy

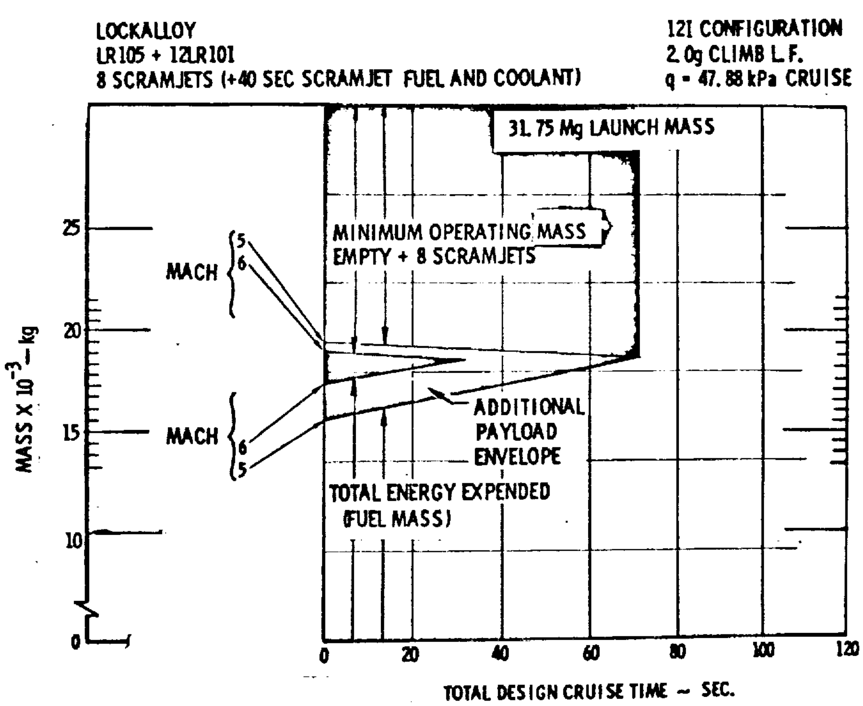


Figure 16 - Rocket Cruise Capabilities - LR-105, 31.75 Mg, 8 Scramjets - Lockalloy

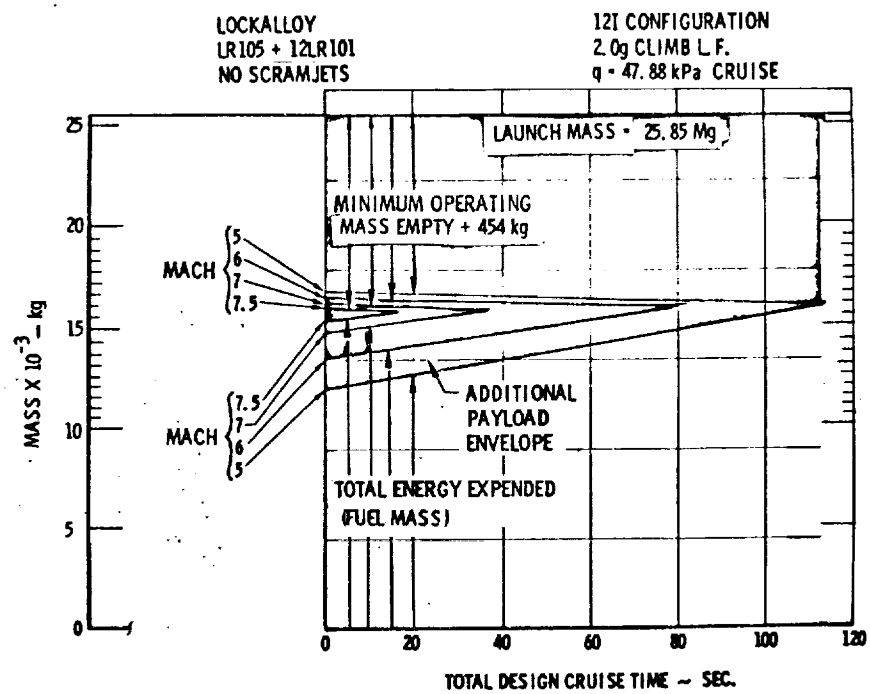


Figure 17 - Rocket Cruise Capabilities - LR-105, 25.85 Mg, No Scramjets - Lockalloy

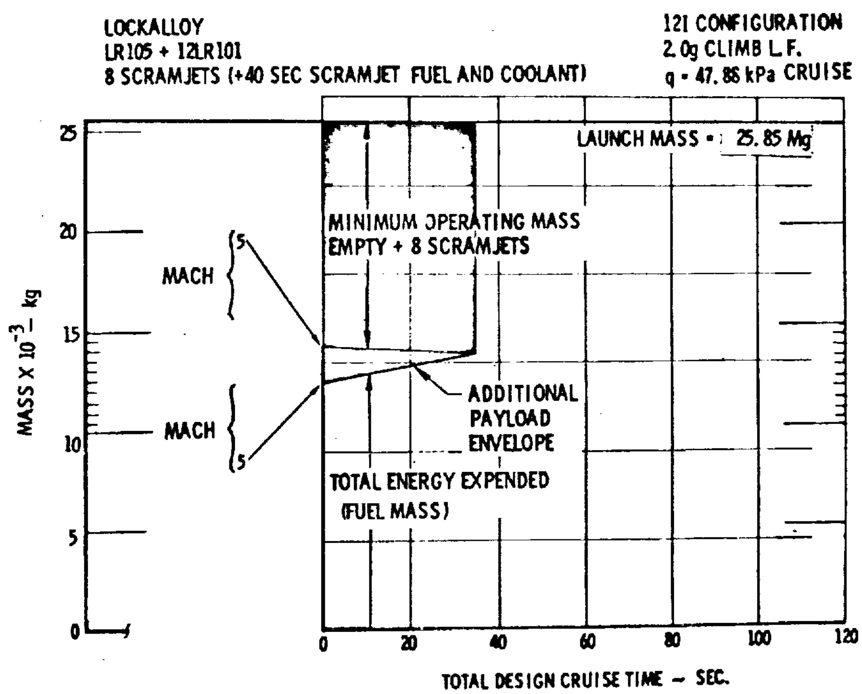


Figure 18 - Rocket Cruise Capabilities - LR-105, 25.85 Mg, 8 Scramjets - Lockalloy

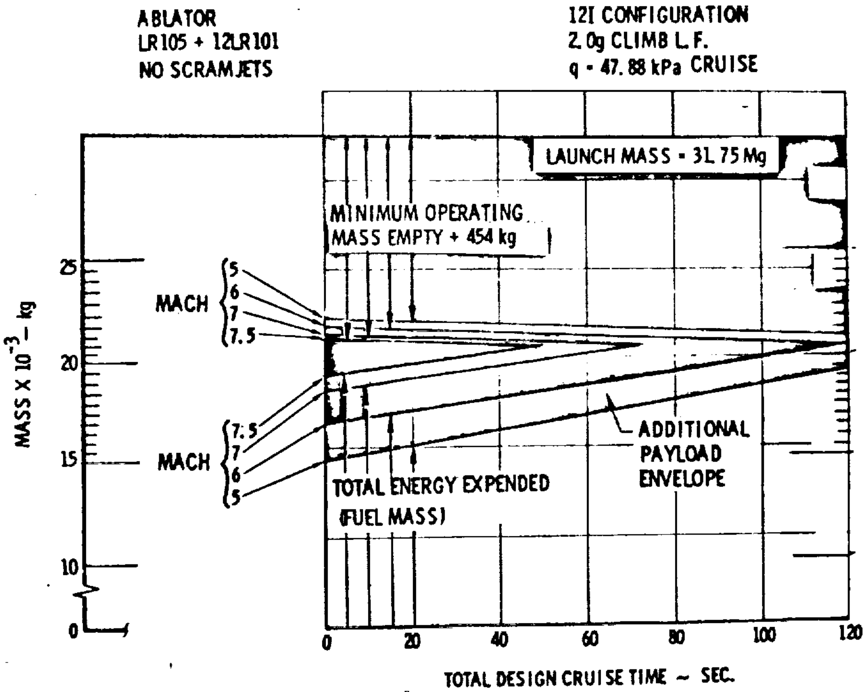


Figure 19 - Rocket Cruise Capabilities - LR-105, 31.75 Mg, No Scramjets - Ablator

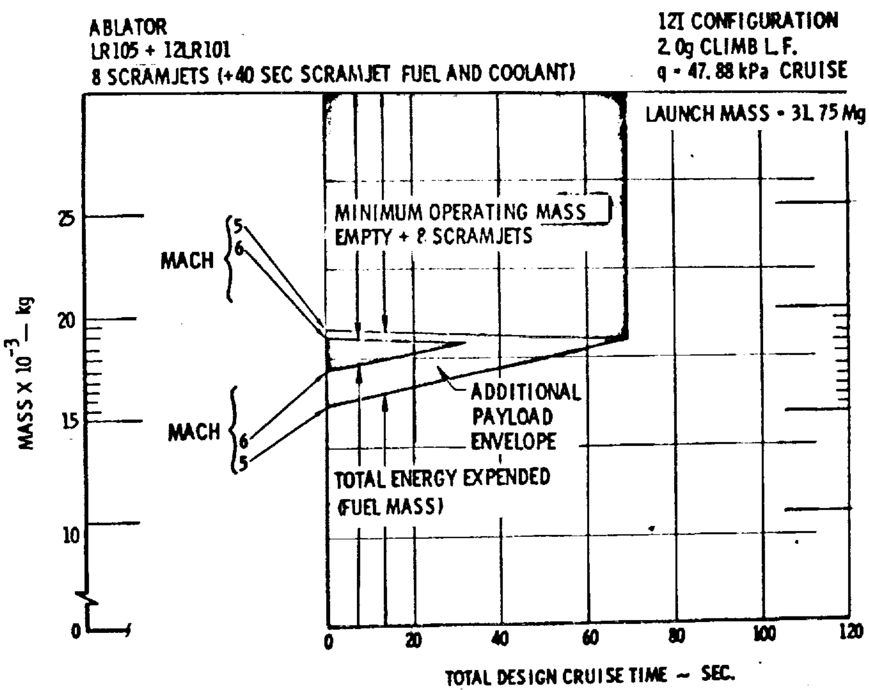


Figure 20 - Rocket Cruise Capabilities - LR-105, 31.75 Mg, 8 Scramjets - Ablator

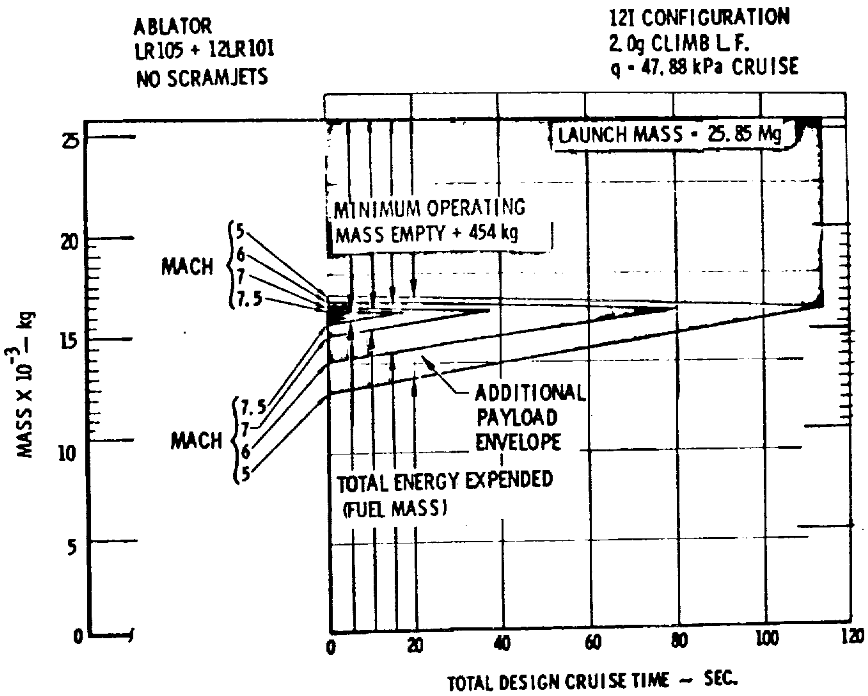


Figure 21 - Rocket Cruise Capabilities - LR-105, 25.85 Mg, No Scramjets - Ablator

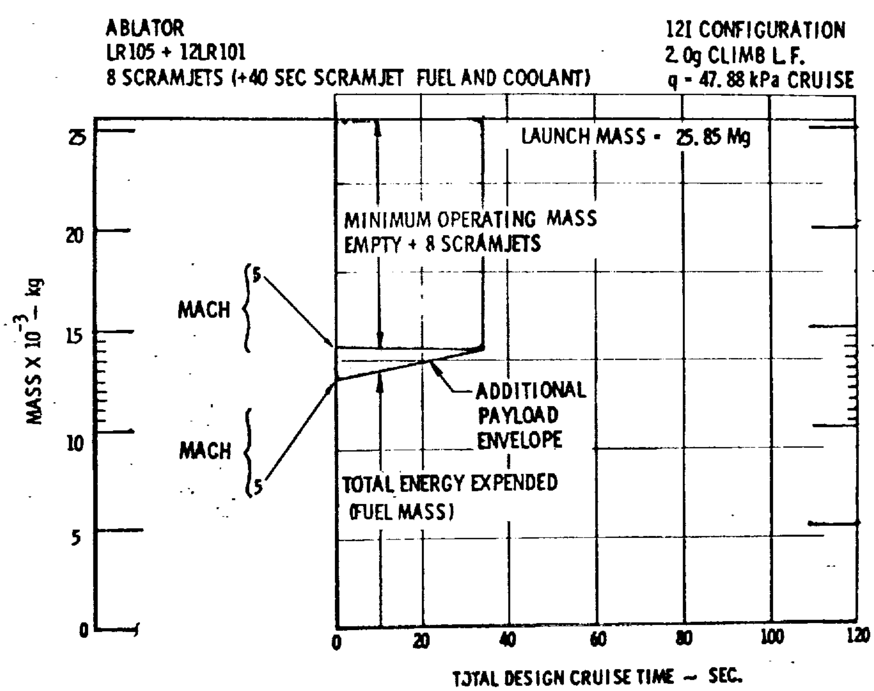


Figure 22 - Rocket Cruise Capabilities - LR-105, 25.85 Mg, 8 Scramjets - Ablator

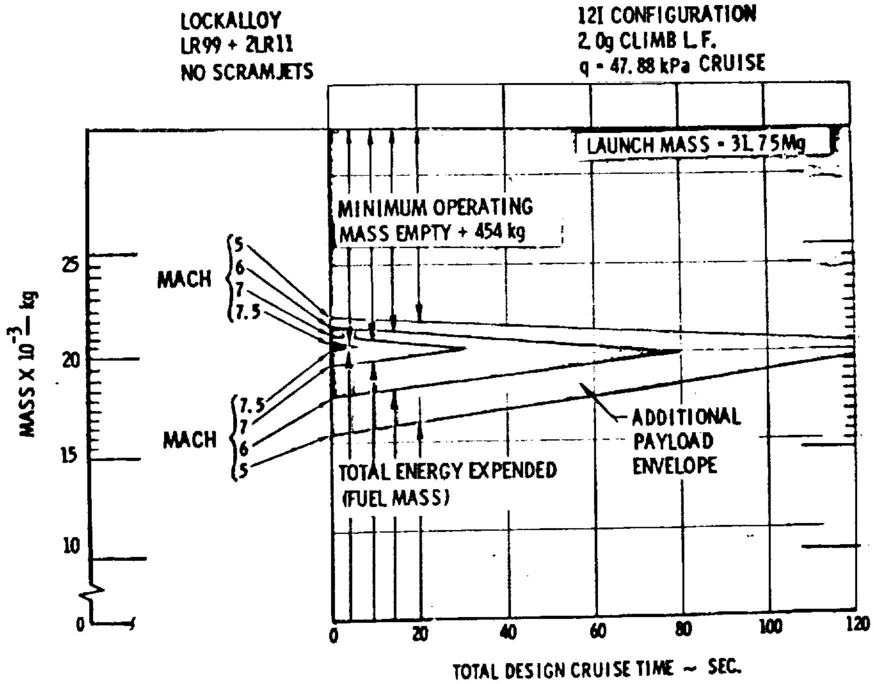


Figure 23 - Rocket Cruise Capabilities - LR-99, 31.75 Mg, No Scramjets - Lockalloy

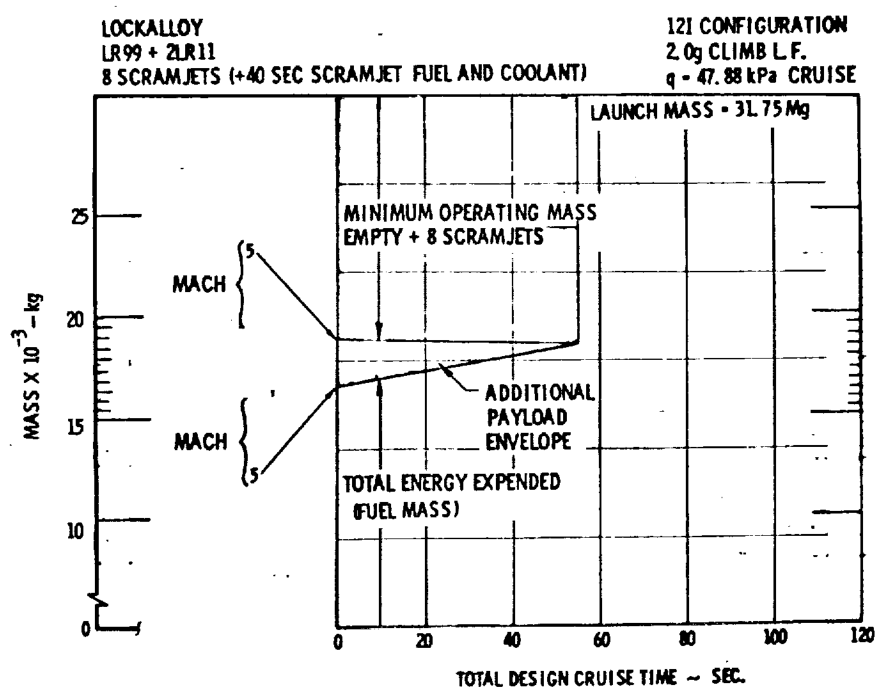


Figure 24 - Rocket Cruise Capabilities - LR-99, 31.75 Mg, 8 Scramjets - Lockalloy

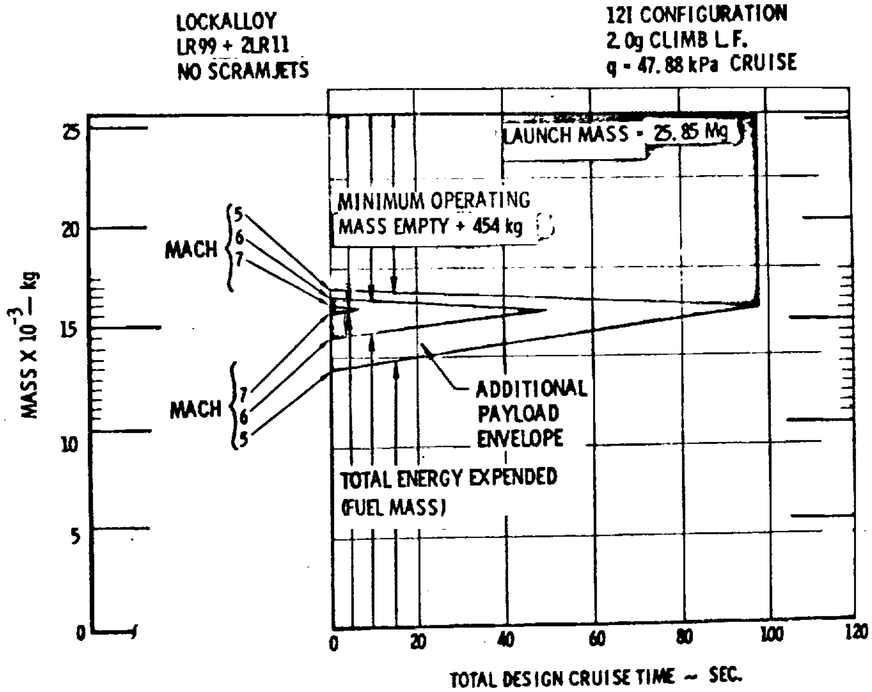


Figure 25 - Rocket Cruise Capabilities - LR-99, 25.85 Mg, No Scramjets - Lockalloy

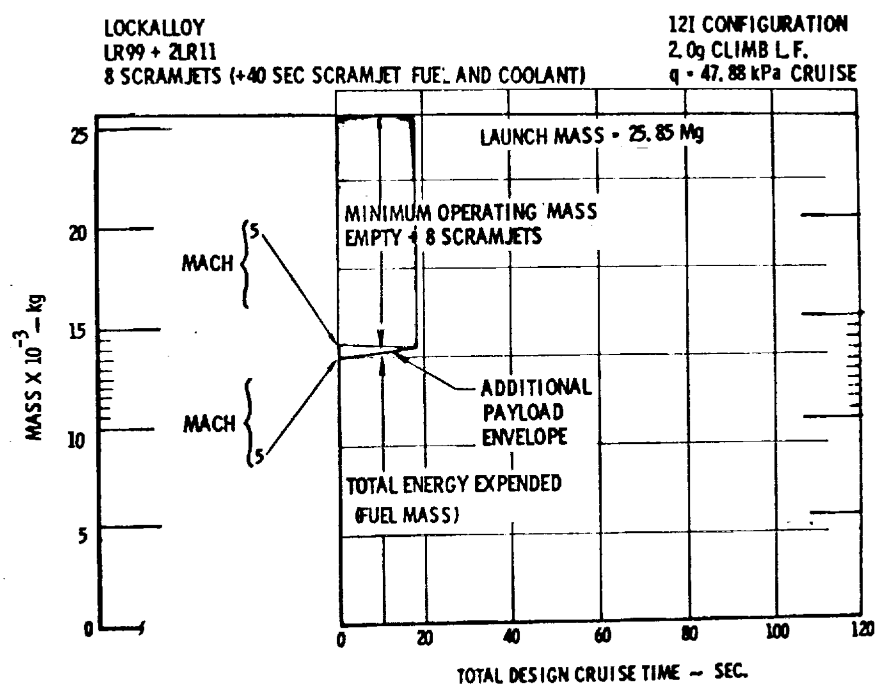


Figure 26 - Rocket Engine Capabilities - LR-99, 25.85 Mg, 8 Scramjets - Lockalloy

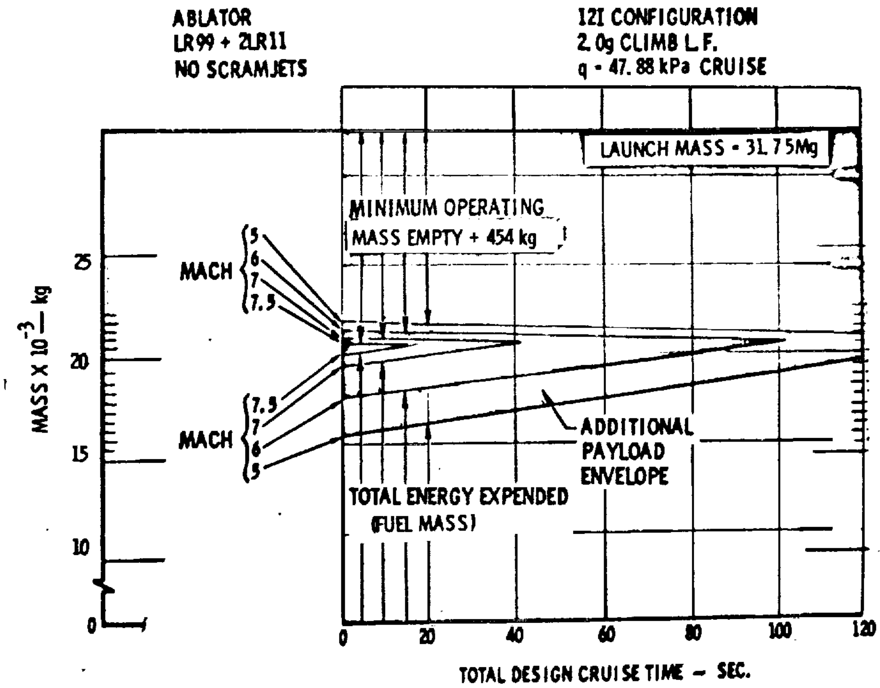


Figure 27 - Rocket Cruise Capabilities - LR-99, 31.75 Mg, No Scramjets - Ablator

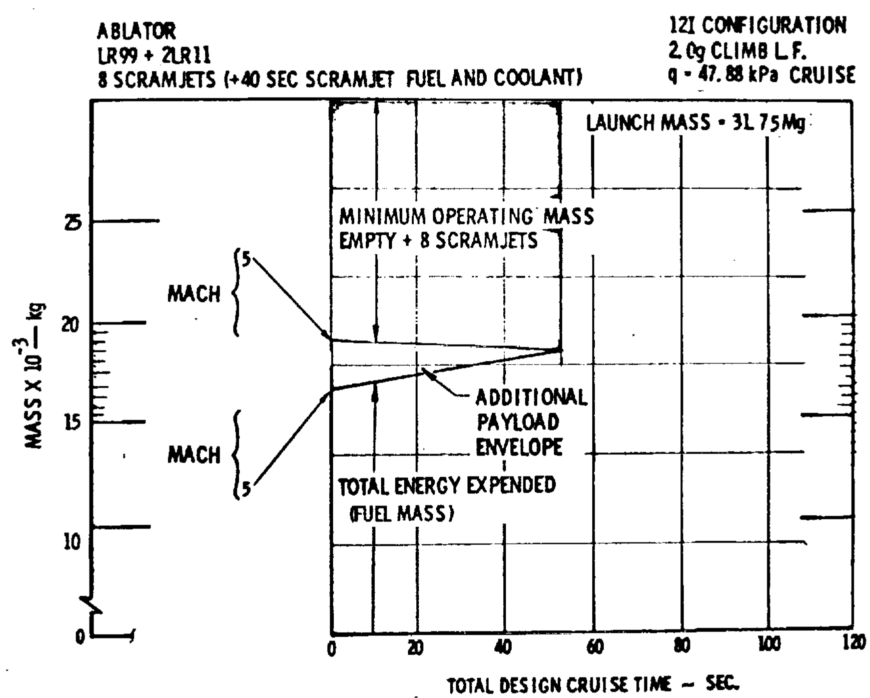


Figure 28 - Rocket Cruise Capabilities - LR-99, 31.75 Mg, 8 Scramjets - Ablator

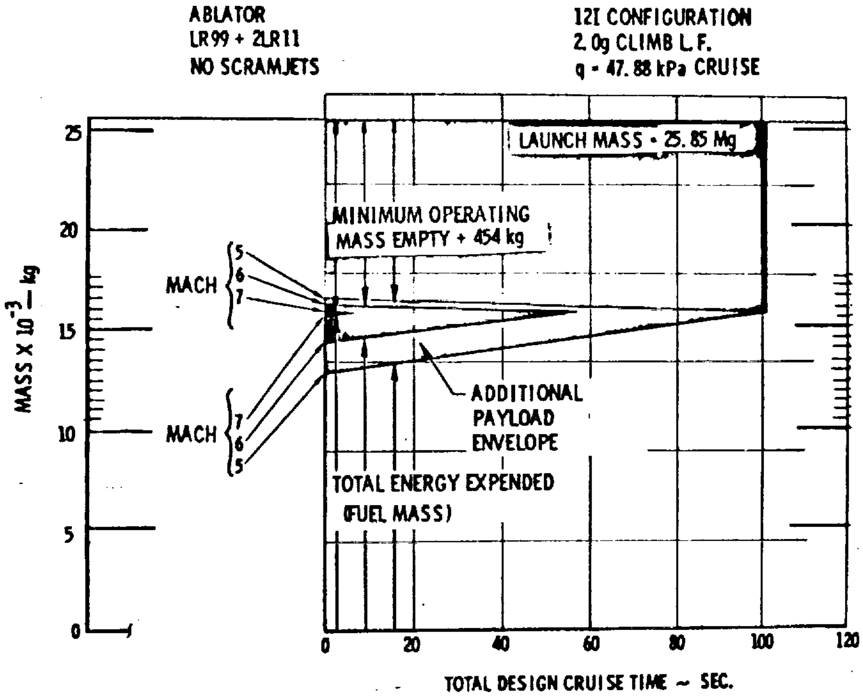


Figure 29 - Rocket Cruise Capabilities - LR-99, 25.85 Mg, No Scramjets - Ablator

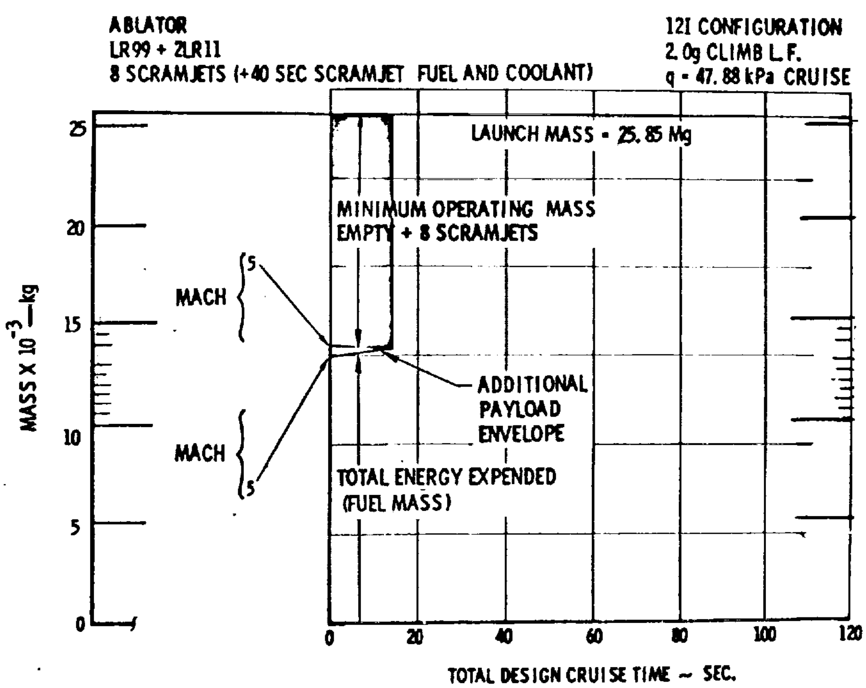


Figure 30 - Rocket Cruise Capabilities - LR-99, 25.85 Mg, 8 Scramjets - Ablator

Consider the vehicle configuration described in Figure 15 as an example. These results indicate that it is possible, within these parameters, to design a vehicle to cruise for 40 seconds at either Mach 5, 6 or 7 with an "Additional Payload" of 553, 331 and 88 kilograms respectively. It also indicates that the maximum design cruise times for Mach numbers 5, 6, 7 and 7.5 to be 120+, 100, 54 and 29 seconds respectively. Figure 15, in comparison to the other vehicle parameter combinations, provide a comprehensive overview as to the design possibilities for a rocket cruise vehicle.

Review of the cruise times results, Figure 15 through 30, yields insight into the design for rocket cruise. The variation of the maximum possible design cruise time (zero additional payload) with Mach numbers is illustrated in the accompanying figures. Regardless of whether the vehicle is equipped with cruise scramjets or not, the 31.75 Mg launch mass family of vehicles exhibits a larger negative trend than the 25.85 Mg group. This indicates decreasing Mach number performance gains with increasing launch mass, as was to be expected. The smaller the design Mach numbers, the larger the difference in design cruise time with increasing launch mass. Its interesting to note that the 25.85 Mg configuration (LR-105 rocket), Figure 21, has greater design cruise time possibilities than the 31.75 Mg configuration (LR-99 rocket), Figure 23, within the higher region of Mach numbers.

Combined with the "maximum possible design cruise time" is the rate change in "Additional Payload" with design cruise time. Analysis has shown that Lockalloy vehicles to have a higher rate change than Ablator TPS vehicles. This tendency poses the question as to whether the LR-105 Lockalloy vehicle could have a greater "Additional Payload" at a conservative design cruise time than the LR-105 Ablator vehicle. Two 31.75 Mg vehicles, Figures 15 and 19, were investigated with results indicating that at Mach 5 and 40 seconds of design cruise, the two configurations to be equivalent in "Additional Payload." With increasing Mach number the greater change in "Additional Payload" of the Lockalloy vehicle is not able to compensate for the large maximum possible design cruise time of the Ablator configured vehicle.

Regardless of the launch mass, the LR-105 plus 12 LR-101 powered Ablator TPS vehicle shows greater design possibilities with respect to rocket cruise and "Additional Payload."

PAYLOAD AND CRUISE TIME VS LAUNCH MASS AND MACH NUMBER

As an additional tool in the investigation, analysis, and recommendation process, data from analysis on "Rocket Cruise Capabilities" and "Variation of Payload with Launch Mass" herein was replotted in Figures 31 through 34. These figures provide further comparison of the design possibilities afforted by each of the configurations.

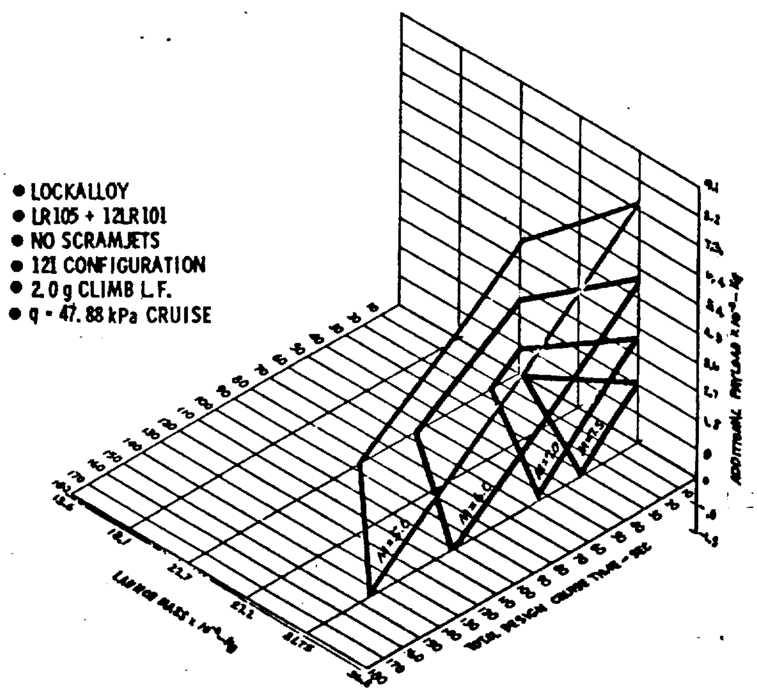


Figure 31 - Payload and Rocket Cruise Time vs Launch Mass and Mach Number - LR-105, No Scramjets - Lockalloy

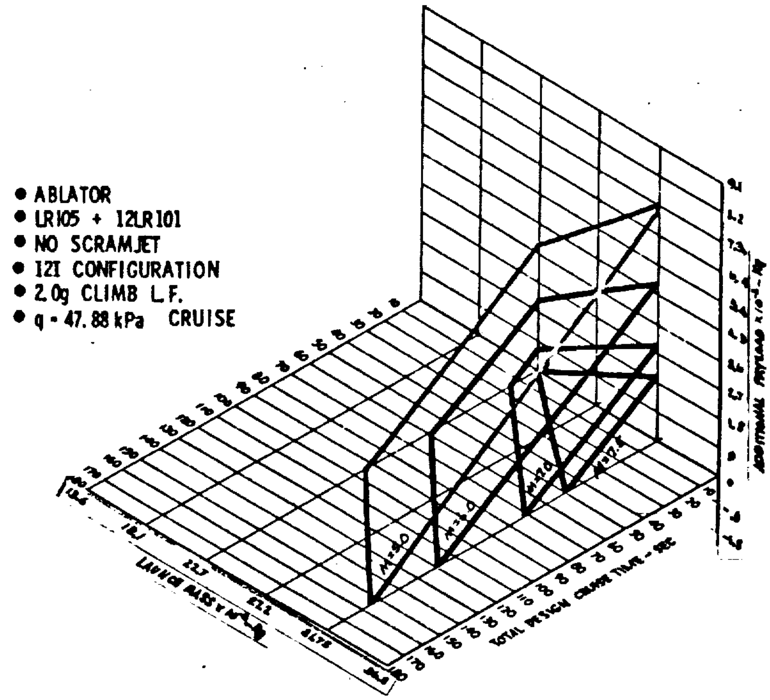


Figure 32 - Payload and Rocket Cruise Time vs Launch Mass and Mach Number - LR-105, No Scramjets - Ablator

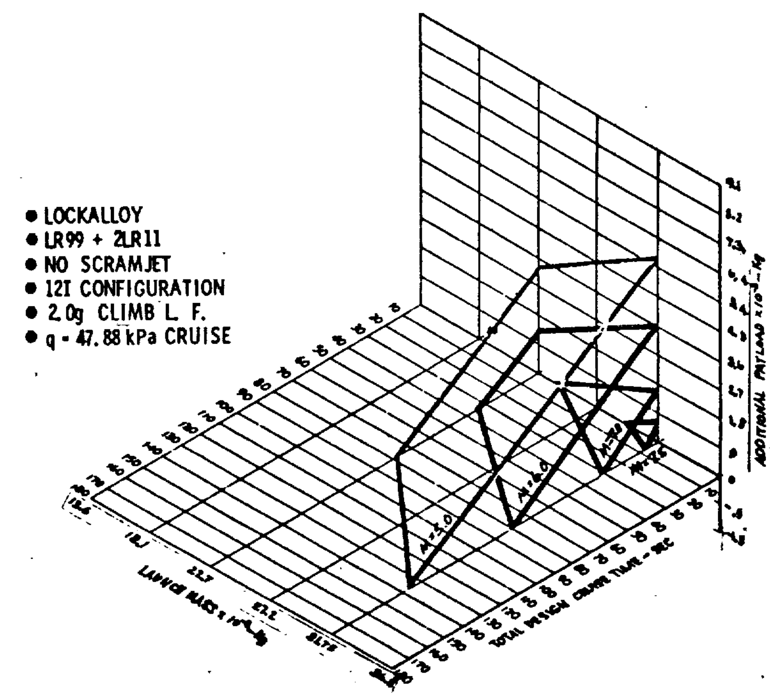


Figure 33 - Payload and Rocket Cruise Time vs Launch Mass and Mach Number - LR-99, No Scramjets - Lockalloy

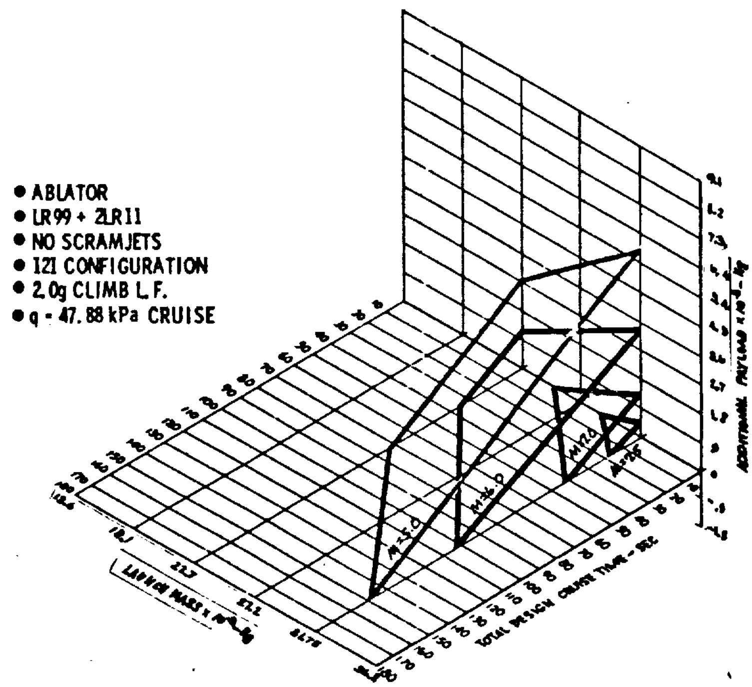


Figure 34 - Payload and Rocket Cruise Time vs Launch Mass and Mach Number - LR-99, No Scramjets - Ablator

VARIATION OF PAYLOAD WITH LAUNCH MASS

Analysis to determine the variation of payload with launch mass relates the summation of the "operating mass empty," "total energy expended" and "payload with launch mass." Figures 35 through 45 present the results of this analysis representing the spectrum of vehicle variables considered, from Mach 6 to 8. Fixed parameters, i.e., vehicle configuration, dynamic pressure at cruise and boost phase maneuver load factor have remained the same as in all investigations. The launch mass is given as the abscissa and each point represents a unique vehicle.

Three separate missions are presented (1) boost to Mach, (2) boost to Mach and cruise for 40 seconds on sustainer engines, and (3) boost to Mach with scramjet cruise for 40 seconds. Each mission was considered independent and any mass segment is not relatable to another.

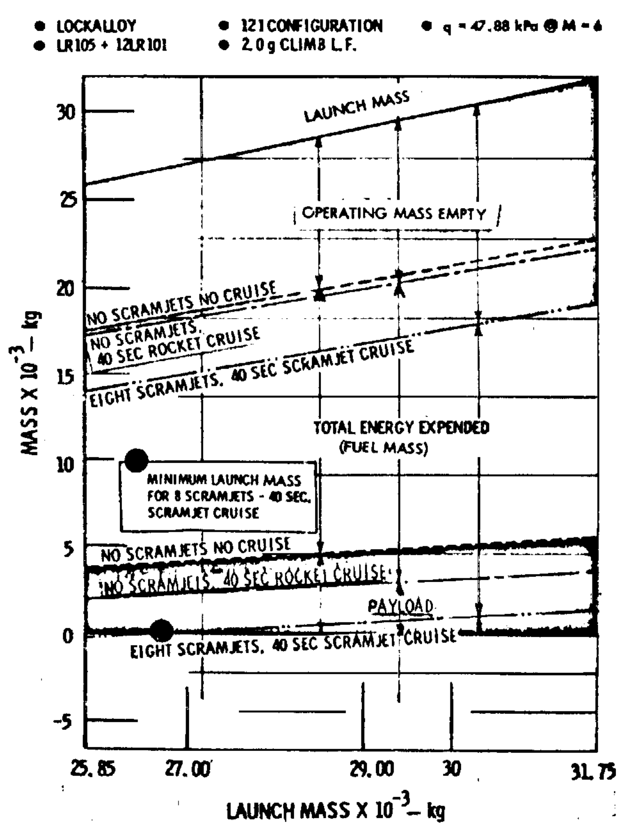


Figure 35 - Variation of Payload with Launch Mass at M = 6, LR-105 - Lockalloy

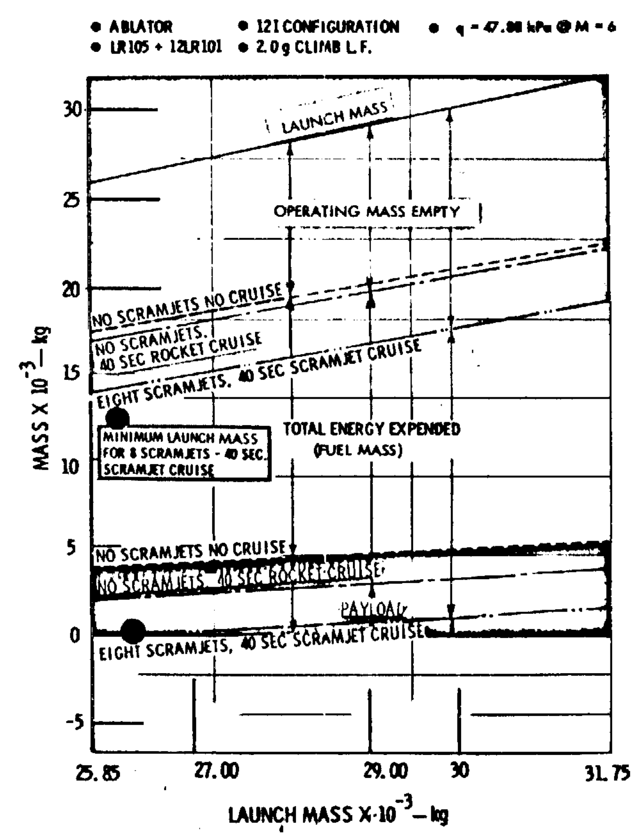


Figure 36 - Variation of Payload with Launch Mass at M = 6, LR-105 - Ablator

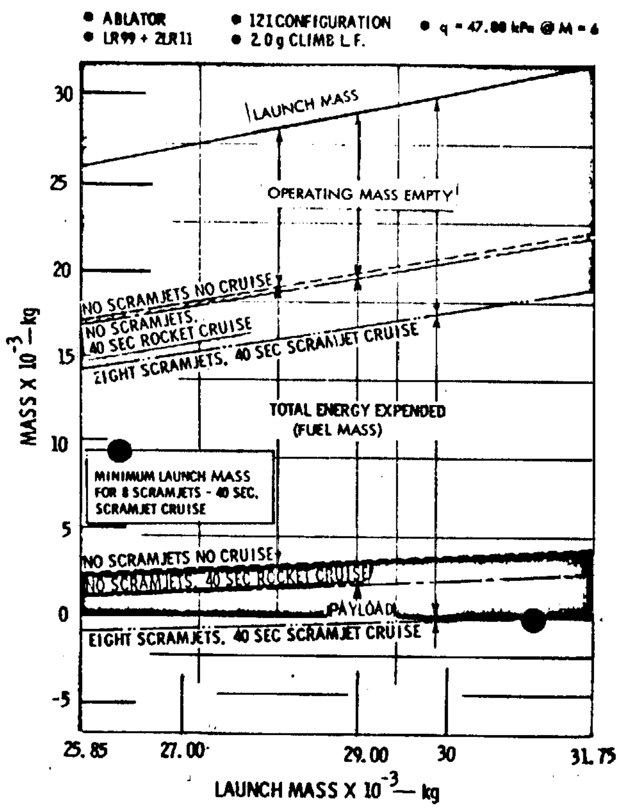


Figure 37 - Variation of Payload with Launch Mass at M = 6, LR-99 - Ablator

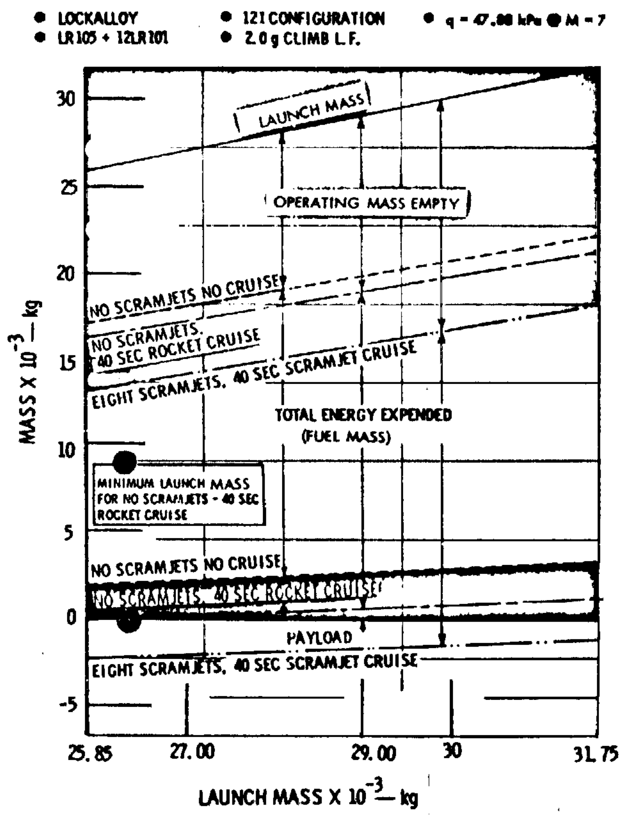


Figure 38 - Variation of Payload with Launch Mass at M = 7, LR-105 - Lockalloy

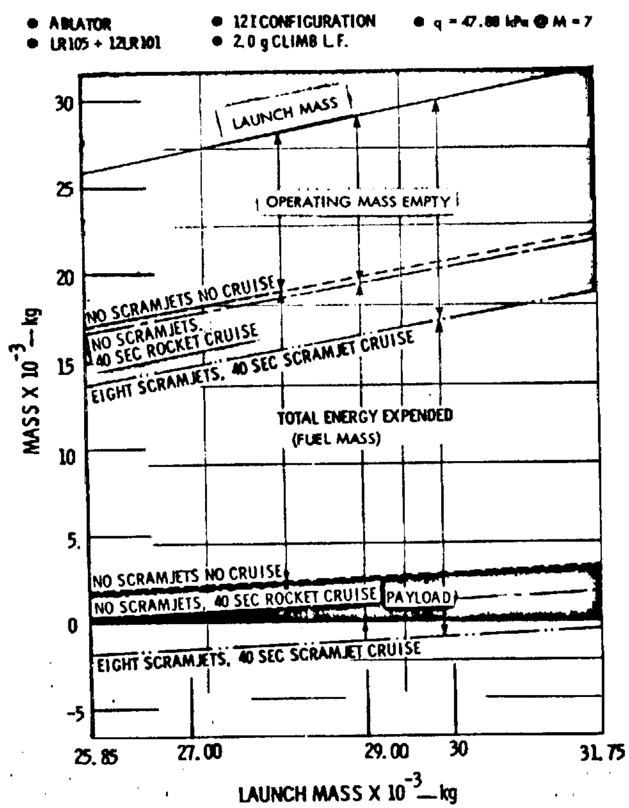


Figure 39 - Variation of Payload with Launch Mass at M = 7, LR-105 - Ablator

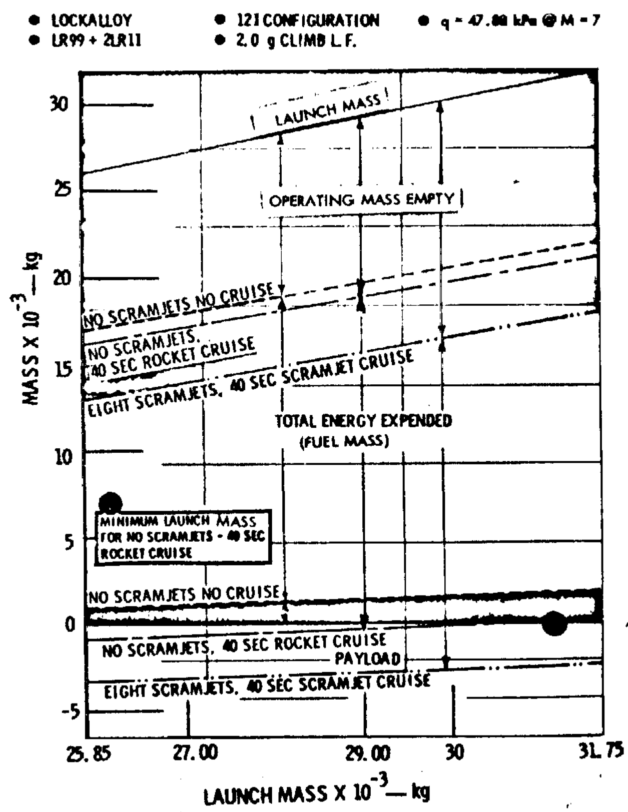


Figure 40 - Variation of Payload with Launch Mass at M = 7, LR-99 - Lockalloy

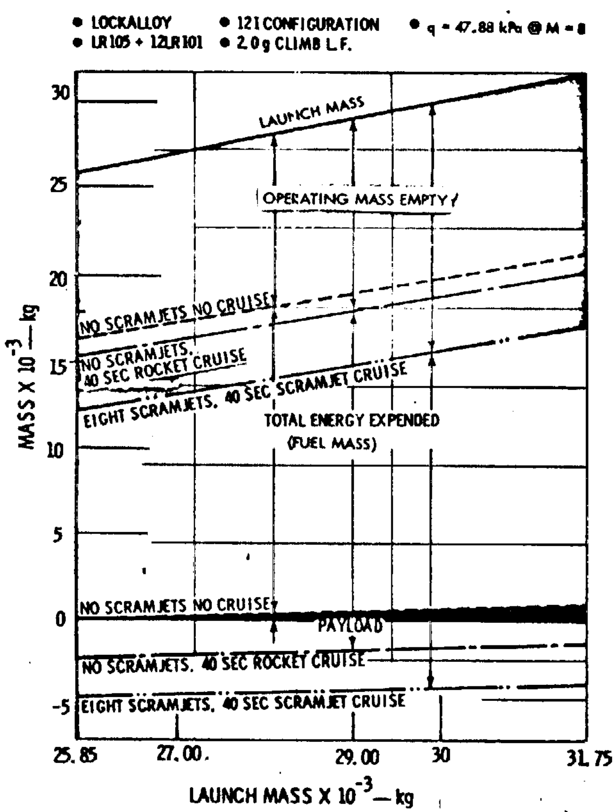


Figure 41 - Variation of Payload with Launch Mass at M = 8, LR-105 - Lockalloy

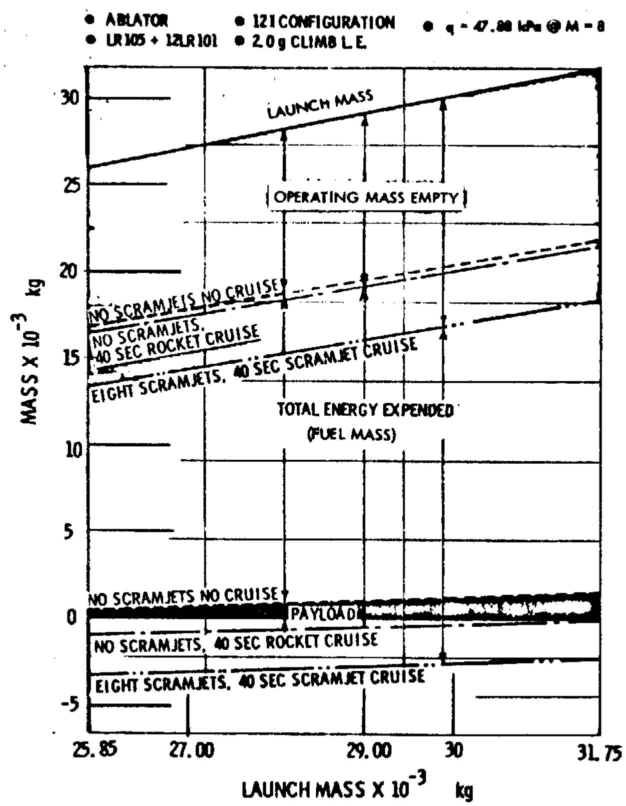


Figure 42 - Variation of Payload with Launch Mass at M = 8, LR-105 - Ablator

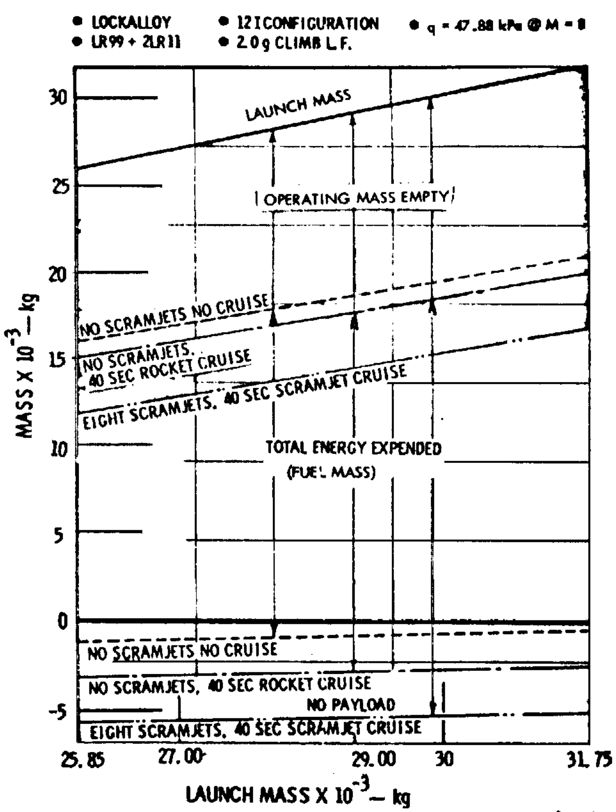


Figure 43 - Variation of Payload with Launch Mass at M = 8, LR-99 - Lockalloy

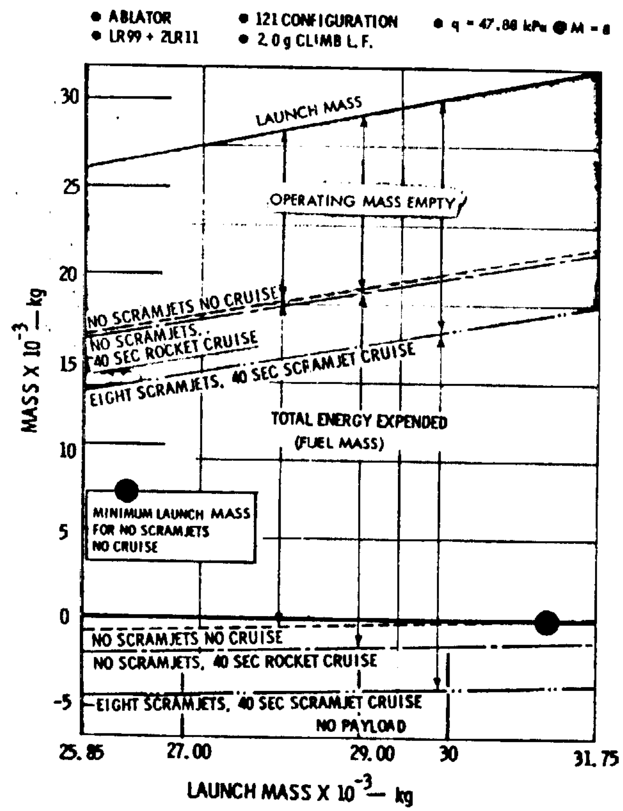


Figure 44 - Variation of Payload with Launch Mass at M = 8, LR-99 - Ablator

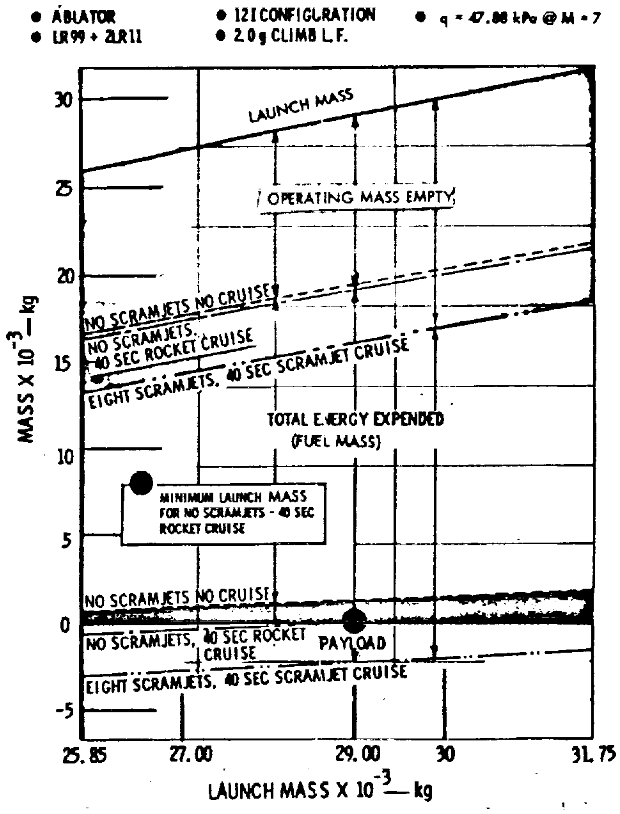


Figure 45 - Variation of Payload with Launch Mass at M = 7, LR-99 - Ablator

Figures were constructed using data developed from other investigations herein, to illustrate the mission design capabilities over a spectrum of launch masses. Assumed linearity of the mission segments was substantiated by the results of aerodynamic data used to support the "mass Analysis" herein. The largest deviation from linearity (approximately 13.6 kilogram) was considered well within the limits of the analytical method used for this analysis.

The intersection of a particular mission's "additional payload" with the abscissa represents that minimum launch mass at which a vehicle can be designed to complete the specified mission. These minimum launch masses have been highlighted by a bold dot on applicable figures.

BOOST FUEL VS MACH NUMBER AND LAUNCH MASS

Figures 46 and 47 represent the results of the investigation to determine the required fuel to boost to a specified Mach number as a function of engine configuration, launch mass and aerodynamic drag configuration. Fixed performance parameters, i.e., dynamic pressure at the end of boost, vehicle configuration and boost phase maneuver load factor, remain constant as with previous analysis.

Figures 46 and 47 represent the "total energy expended" obtained from the computer program of the boost phase for the launch mass of 25.85 and 31.75 Mg. Launch mass data, other than those investigated in the computer program were established by linear interpolation. The mechanics of the computer program are covered in the Mission Profile Section of the Phase I Study report, Reference 1.

Comparison of the two figures indicates a fuel consumption savings of between 5 to 7 percent for the LR-105 plus 12 LR-11 engine combinations as compared to the LR-99 plus 2 LR-11 combination.

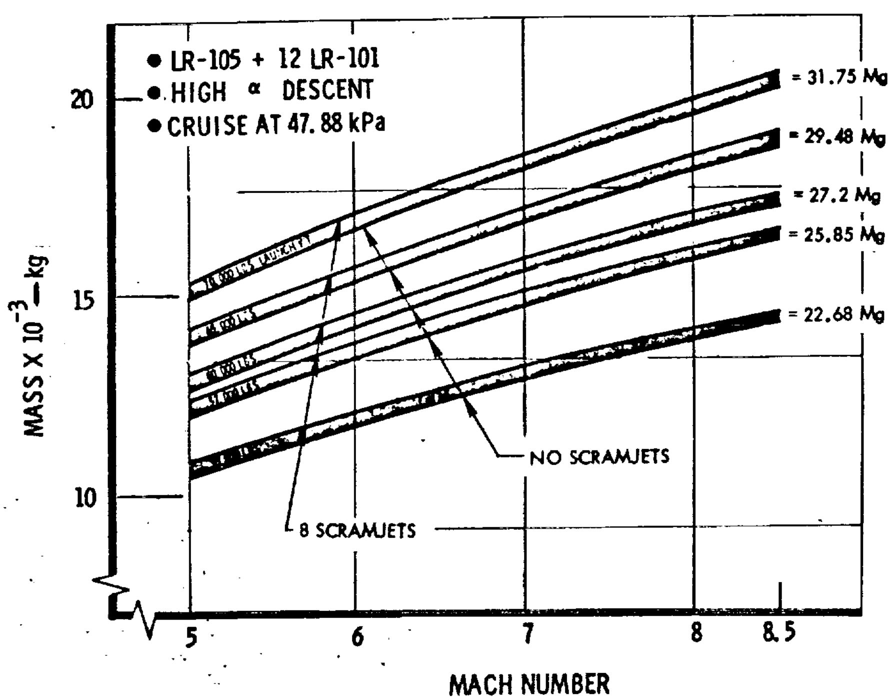


Figure 46 - Boost Fuel vs Mach Number and Launch Mass - LR-105

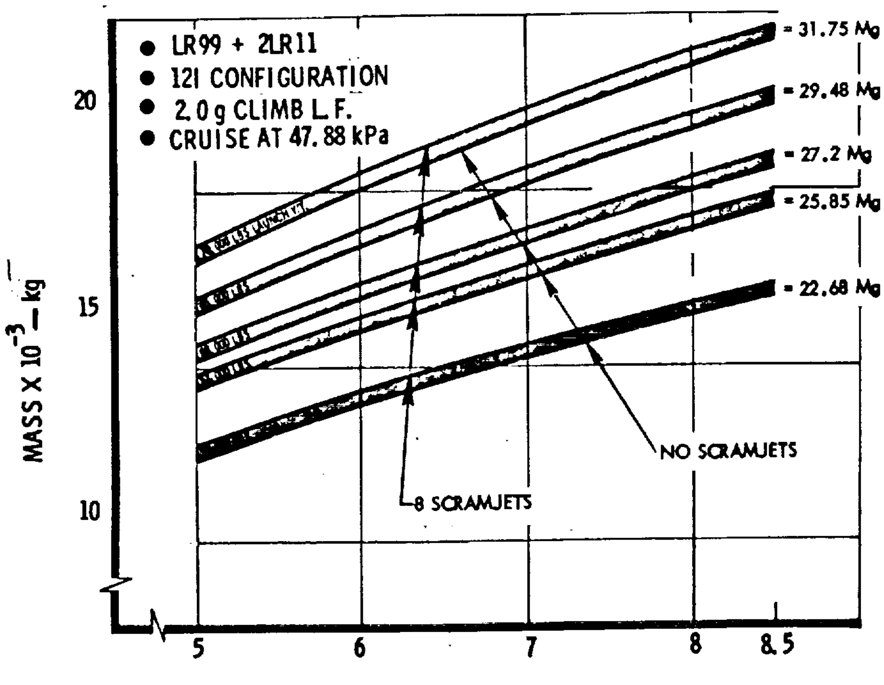


Figure 47 - Boost Fuel vs Mach Number and Launch Mass - LR-99

THERMAL PROTECTION SYSTEM MASS INCREMENTS

A parametric study of the Thermal Protection System (TPS) was conducted of the effect of cruise Mach number, cruise time, vehicle launch mass and dynamic pressure at cruise on the total Ablator and Lockalloy System mass. Transient thermal analysis of vehicle surface had to be made for each different vehicle trajectory, as was done in the Phase I Study, Reference 1 - THERMAL ANALYSIS section.

Lockalloy Thickness Analysis - In order to reduce the work to a reasonable magnitude, the number of vehicle locations analyzed was reduced from that used in the Phase I study. Representative locations over the vehicle were chosen and plots of Lockalloy peak temperature versus Lockalloy thickness were prepared for each of these locations, and for the range of parameters shown in Figure 48. Lockalloy thicknesses which yield 589 K maximum could then be determined for each of the analyzed vehicle locations, and at any of the established flight

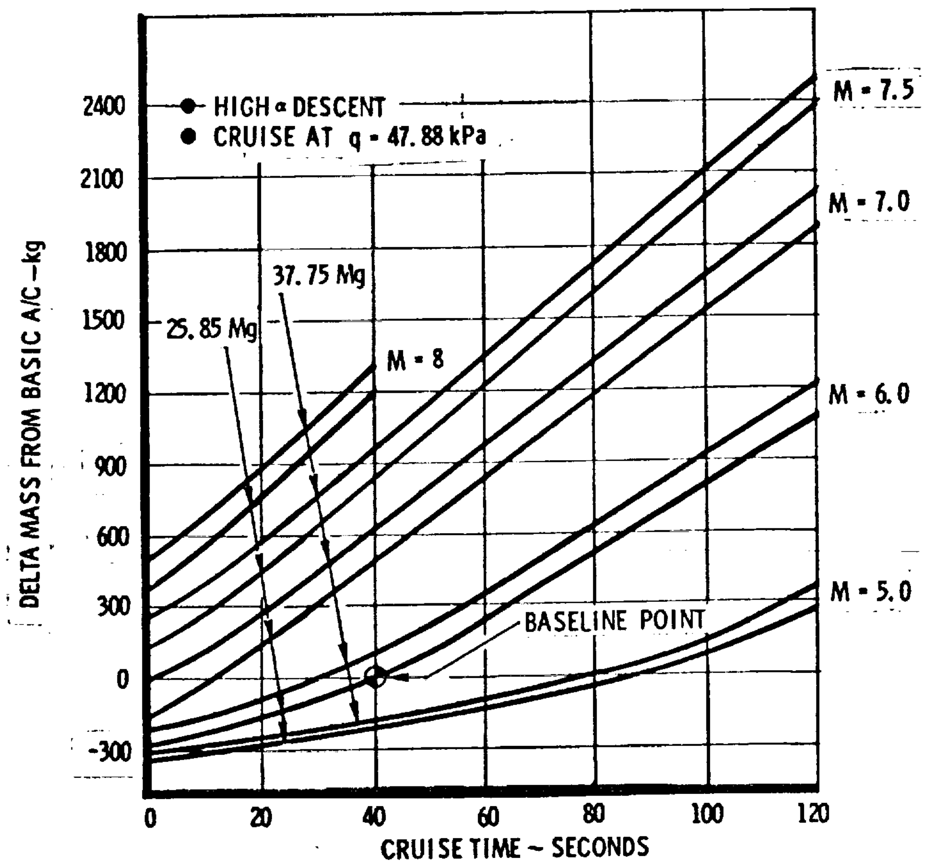


Figure 48 - TPS Mass Increment - Lockalloy

conditions. Corresponding thickness for other locations on the vehicle were then approximated by ratioing the thicknesses determined in the Phase I study, Figure 104 and 105 of Reference 1, up or down by the same percentage that the parametrically studied point changed at a similar location. For instance, if for a given set of flight conditions different from those used in the Phase I study, the thickness of a point on the lower surface increased by 20% from the baseline condition, then all lower surface points were assumed to increase by 20% for the same flight condition.

This technique yielded a sufficient amount of TPS requirement data to support overall vehicle parametric studies. The data, in the form of the thickness vs maximum temperature curves was used to assess the actual Lockalloy thickness requirements over the vehicle for each flight/vehicle condition. Actual thicknesses were determined as described in the Thermal Analysis of Reference 1, from a combination of thermal protection, minimum gauge and minimum structural requirements. The set of Lockalloy thicknesses that were produced were related to appropriate vehicle areas to provide incremental mass changes from the basic vehicle/mission (Baseline Point). These delta masses are reflected in Figure 48 for the range of parameters studied.

Ablator TPS Thickness Analysis - Ablator TPS analysis was conducted using the same method employed for the Lockalloy heat-sink system described above. Thicknesses produced by this analysis were related to appropriate vehicle areas, defined and established in the Phase I Study reported in Reference 1. This provided incremental changes from the basic vehicle/mission (Baseline Point) reflected in Figure 49 for the range of parameters studied.

Ablator/Reuseable Surface Insulation Distribution - Ablator TPS analysis described above, produced external surface temperature plots for the conditions analyzed from which it was possible to show the amount of Ablator and RSI that each set of conditions created.

Figure 50 depicts the conditions and results of the X-24C Phase I Study, with Figures 51 and 52 depicting the increase in RSI required as cruise speed Mach numbers increased.

Previous X-24C studies, reported in Reference 2 and 3, substantiated use of Ablator TPS up to temperatures of 922 K. Reports further indicate testing above 922 K but of varying Ablator characteristics and an indication that the characterization of material at the higher temperatures was required. To avoid undue risk and higher X-24C development costs this study has used the 922 K temperature as the Ablator cutoff, Figure 53, in lieu of the 1033 K temperature cutoff used in the Phase I Study. Figures 54 and 55 depict the increase in RSI required as cruise speed Mach numbers increase based on the 922 K cutoff.

The use of RSI described under this analysis does not include additional RSI usage described separately under "INSULATOR TPS PROBLEMS" herein.

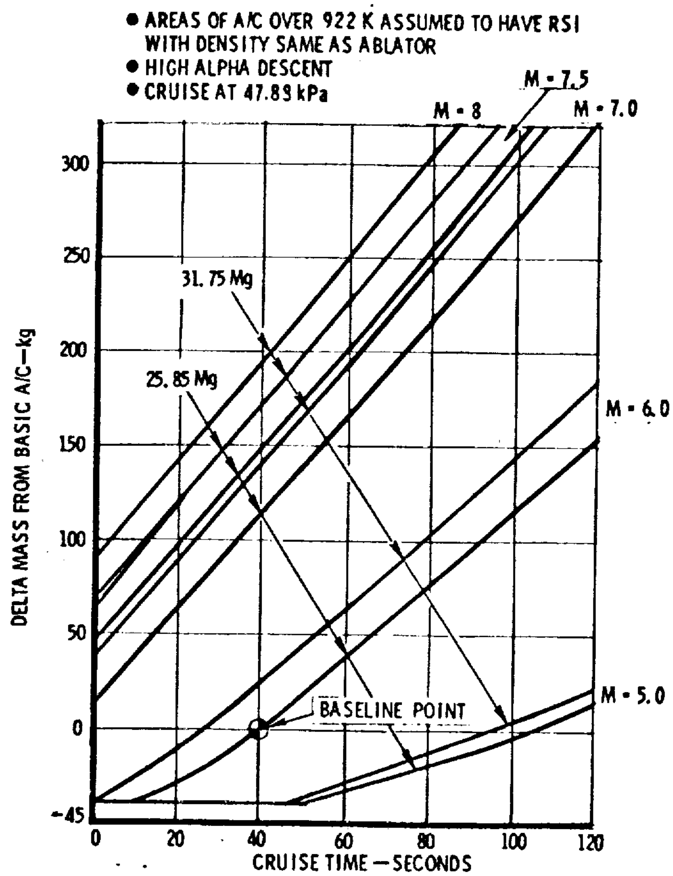


Figure 49 - TPS Mass Increment - Ablator

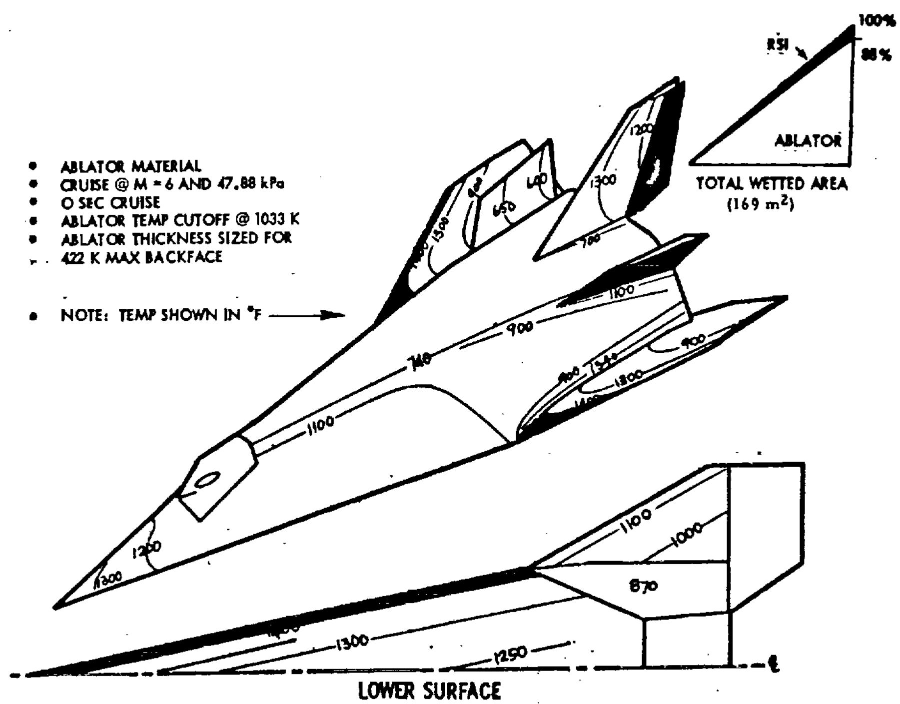


Figure 50 - Ablator Surface Area - Mach 6/1033 K

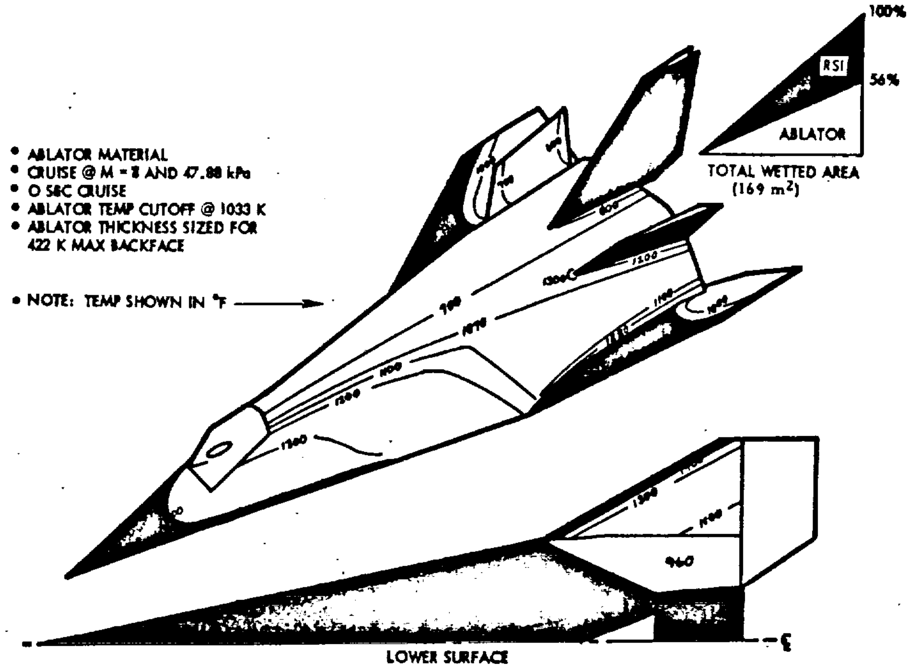


Figure 51 - Ablator Surface Area - Mach 7/1033 K

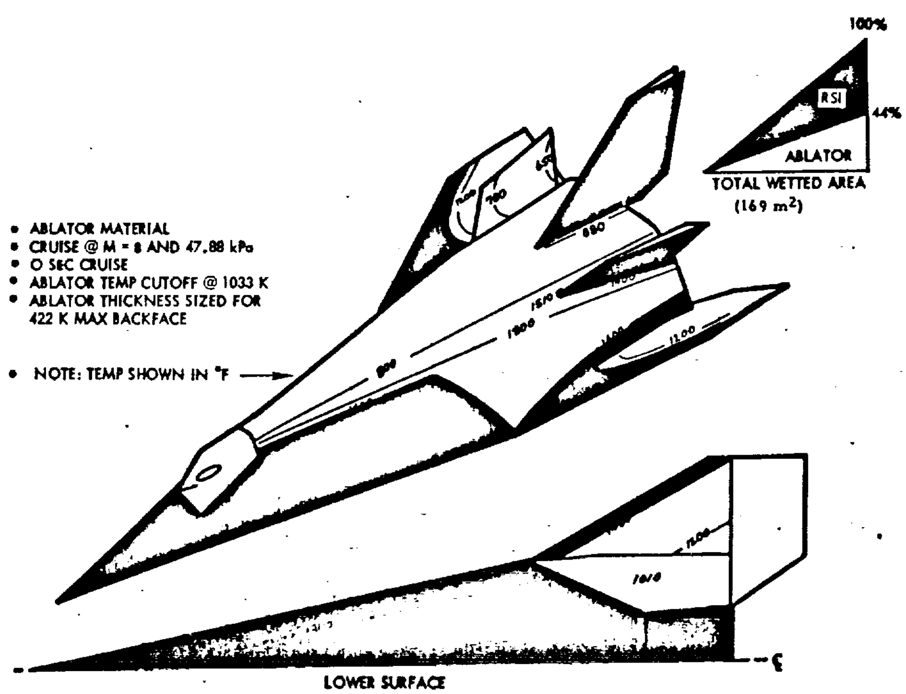


Figure 52 - Ablator Surface Area - Mach 8/1033 K

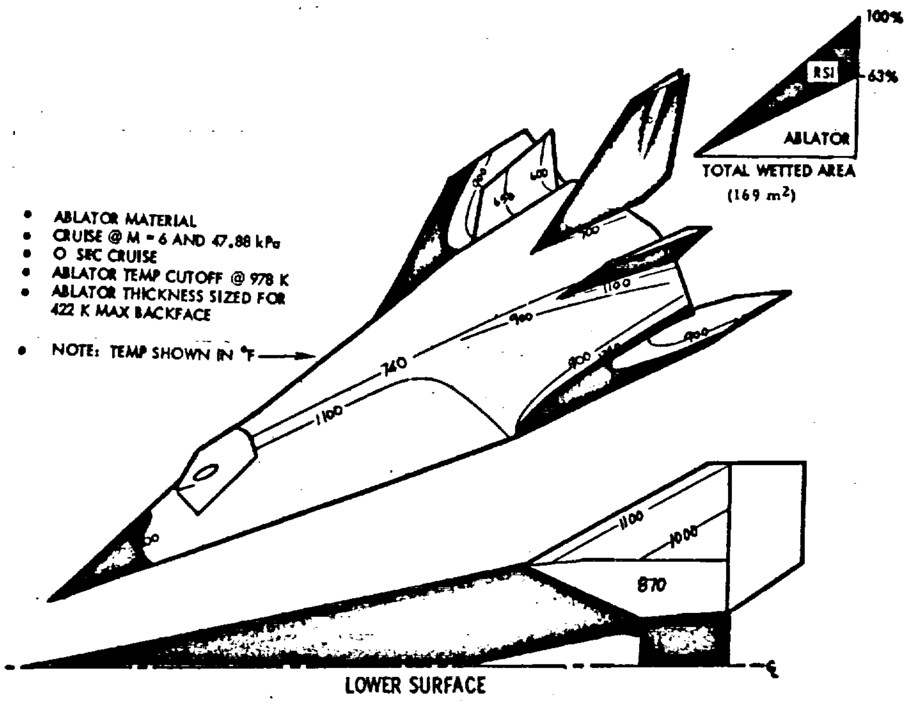


Figure 53 - Ablator Surface Area - Mach 6/922 K

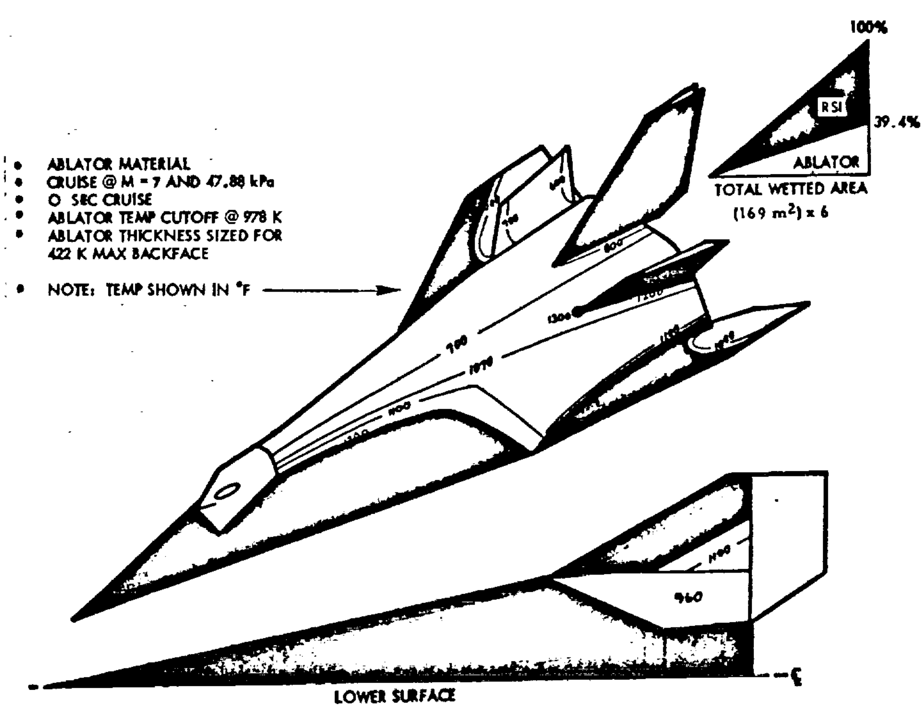


Figure 54 - Ablator Surface Area - Mach 7/922 K

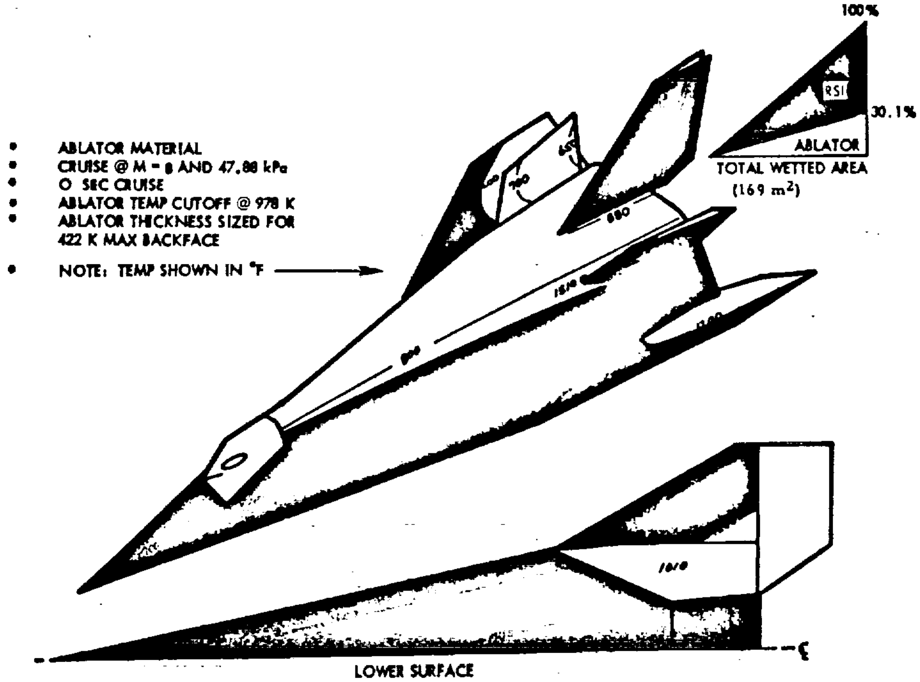


Figure 55 - Ablator Surface Area - Mach 8/922 K

OFF DESIGN CAPABILITIES

The "off design capabilities" of two vehicle arrangements were investigated to determine the parameters limiting cruise potential. Each configuration investigated utilize the LR-105 plus 12 LR-101 engine combination and the Lockalloy heat-sink structure. To illustrate cruise potential, cruise time was selected as the dependent variable with Mach number, rocket fuel, TPS and launch mass as the independent variables. The off design launch mass of the "design point vehicles" were also investigated.

A design point vehicle of 31.75 Mg launch mass, designed to cruise for 40 seconds on scramjets was selected. The second vehicle had a launch mass of 29.03 Mg, but with a structural capability of 31.75 Mg, and was to achieve the maximum Mach number possible without cruise. The first of these two vehicles had been partially investigated as part of the analysis described in the "MAXIMUM MACH NUMBER ATTAINABLE" herein.

Figure 56 reflects the results of the investigation of the 31.75 Mg vehicle and can be interpreted as follows; the design point is highlighted by a bold dot at the intersection of heat-sink limit, fuel cruise time, and maximum Mach number attainable. Off design capabilities are indicated by the zone titled "capability with scramjets." Cruise time capability was found to be bounded by rocket fuel capacity below Mach 5.76. From Mach 5.76 to 6.56 the heat sink capability of the vehicle structure limits the amount of cruise time available. As an example consider the mission to cruise at Mach 6. This vehicle has the capability as an 'off design' mission to boost to Mach 6, level off and cruise for 25 seconds on sustainer engines and then continue the cruise on scramjets for a total of 63 seconds.

Interpretation of the Figure 57 investigation results of the 29.03 Mg vehicle is made in the similar manner as noted above for Figure 56.

The "off design capabilities," Figures 56 and 57 were determined by the following inputs:

(1) Off design heat sink cruise limits were determined from the TPS mass increments presented in Figure 48.

(2) Knowing the "design point vehicles" TPS mass the design cruise limit was calculated assuming linearity between Mach numbers and in addition also calculated assuming launch mass linearity.

(3) Rocket fuel cruise limits were calculated knowing the total fuel capacity of the "point design vehicle."

(4) Figure 46 was used to establish the quantity of fuel required to boost to Mach which was then subtracted from the fuel total to establish fuel quantity available for cruise, and

(5) Using the sustainer engines Isp established the cruise duration.

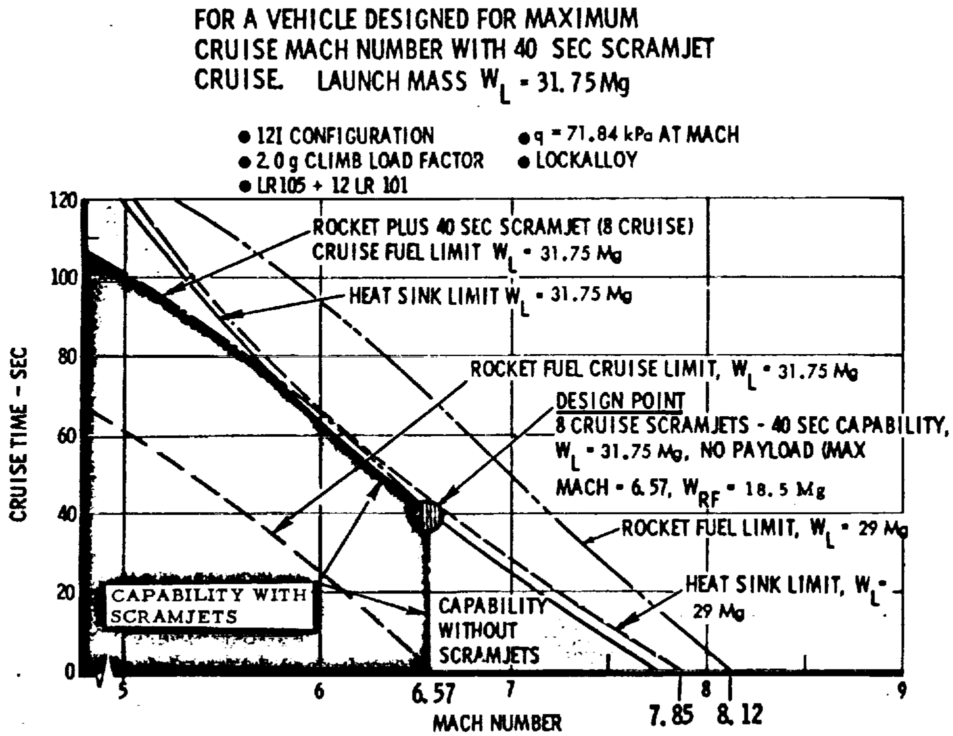


Figure 56 - Off Design Capabilities - Design for Scramjet Cruise

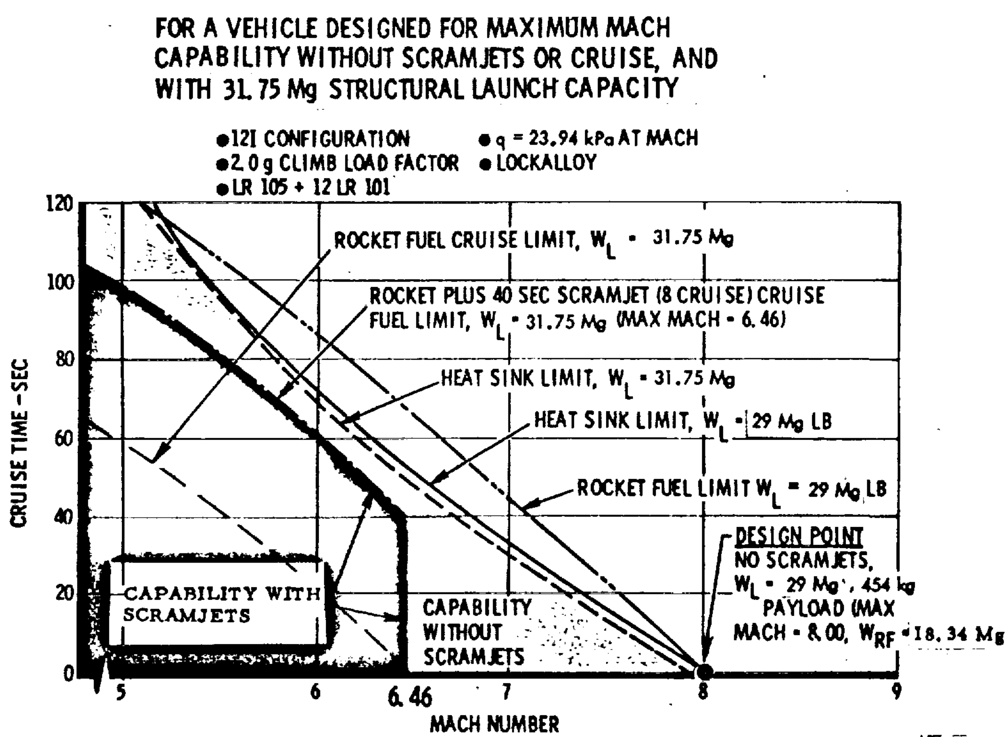


Figure 57 - Off Design Capabilities - Design with No Scramjet or Cruise

Using the same procedures noted above, to investigate the two vehicles reflected in Figures 56 and 57, fourteen different vehicles configurations were analyzed. Half of these vehicles were considered limited by the B-52 maximum launch capacity while the remaining vehicles were specified to perform a specific mission at a minimum launch mass. All engine and TPS variations were considered. Table 2 reflects the results of this investigation on the fourteen vehicles in addition to the two addressed in Figures 56 and 57, and identified as item 1 and 2 in the table. In addition to the basic design condition for each vehicle the table also summarizes the "off design" potential of each vehicle as well as the cost of each configuration.

Within the first eight configurations on Table 2 the odd numbered items represent configurations designed at the maximum launch mass of 31.75 Mg and cruise for 40 seconds at the highest possible Mach number on scramjets. For the "off design" capability the scramjets are removed and various mission limits, without scramjets, were established and tabulated. The even numbered items represent vehicles designed to boost to Mach without cruise. However, the structural capacity of 31.75 Mg was included to meet the "off design" scramjet cruise requirement.

The second eight configurations shown on Table 2 represent vehicles designed to perform a specific mission. Odd number configurations (items) were investigated to determine the minimum launch mass to perform a mission to cruise for 40 seconds on scramjets at Mach 6. (Item 11 required the maximum launch mass capacity of the B-52 to reach slightly under the Mach 6 requirement.) Similarily to the first eight configurations the even numbered vehicles in the second eight configurations represent vehicles in the "off design" condition having the same structural capacity of the preceding odd number configuration.

Generally those configurations in the "off design" condition without scramjets are limited by the thermal protection system. Whether it be boost to the maximum Mach attainable or cruise for 40 seconds on sustainer engines the TPS was found to be the limiting factor. The opposite is the rule for configurations,

DESIGN CONDITION											OFF DESIGN CAPABILITY										
														TPS Limited △8		Rocket Limited △5			Cost – M$ January 1976 Dollars		
Vehicle Configuration	Launch Mass kg x 10^{-3} (lb x 10^{-3})	Engine	TPS	Scramjets	Cruise Mach	Cruise On Rockets	Time On Scramjets	Additional Payload kg (lb)	Operating Mass Empty kg x 10^{-3} (lb x 10^{-3})	Fuel Capacity kg x 10^{-3} (lb x 10^{-3})	Launch Mass kg x 10^{-3} (lb x 10^{-3})	Scramjets	Additional Payload kg (lb)	40 Sec Cruise TPS - M	0 Sec Cruise TPS - M	With Scramjets △7	Without Scramjets △6	Without Scramjets △7	One Vehicle	Two Vehicles	Two Vehicles + Initial Spares, AGE, Data
1	31.75 (70)	△1	△3	8 (Cruise)	6.57	0 Sec	40 Sec	0	13.24 (29.2)	18.5 (40.8)	29 (64)	0	454 (1000)	6.67	7.85	-	8.12	7.28	44.7	59.0	63.4
2	△9 29 (64)	△1		None	8.00		0 Sec	454 (1000)	10.2 (22.55)	18.35 (40.45)	31.75 (70)	8 (Cruise)	0	6.72	7.90	6.46	-	-	43.9	57.9	62.3
3	31.75 (70)	△2		8 (Cruise)	5.98		40 Sec	0	13.06 (28.8)	18.69 (41.2)	29 (64)	0	454 (1000)	6.10	7.24	-	7.47	6.72	43.9	57.9	62.3
4	△9 29 (64)	△2	△3	None	7.38		0 Sec	454 (1000)	10.1 (22.3)	18.46 (40.7)	31.75 (70)	8 (Cruise)	0	6.12	7.26	5.83	-	-	43.6	57.6	61.9
5	31.75 (70)	△1	△4	8 (Cruise)	6.80		40 Sec	0	12.88 (28.4)	18.87 (41.6)	29 (64)	0	454 (1000)	6.95	7.12	-	8.42	7.47	48.0	63.8	69.5
6	△9 29 (64)	△1		None	8.38		0 Sec	454 (1000)	9.9 (21.9)	18.64 (41.1)	31.75 (70)	8 (Cruise)	0	7.90	8.13	6.65	-	-	48.2	64.3	70.1
7	31.75 (70)	△2		8 (Cruise)	6.07		40 Sec	0	12.93 (28.5)	18.82 (41.5)	29 (64)	0	454 (1000)	6.36	6.52	-	7.58	6.32	47.1	62.5	68.3
8	△9 29 (64)	△2	△4	None	7.45		0 Sec	454 (1000)	9.9 (21.9)	18.64 (41.1)	31.75 (70)	8 (Cruise)	0	7.11	7.28	5.95	-	-	47.4	63.1	68.3
9	26.76 (59)	△1	△3	8 (Cruise)	6.00		40 Sec	0	12.1 (26.7)	14.65 (32.3)	24 (53)	0	454 (1000)	6.12	7.29	-	7.70	6.64	39.6	52.3	56.2
10	△10 24 (53)	△1		None	7.54		0 Sec	454 (1000)	9.3 (20.25)	14.4 (31.75)	26.76 (59)	8 (Cruise)	0	6.25	7.42	5.86	-	-	39.3	51.9	55.8
11	31.75 (70)	△2		8 (Cruise)	5.98		40 Sec	0	13.06 (28.8)	18.69 (41.2)	29 (64)	0	454 (1000)	6.10	7.24	-	7.47	6.72	43.9	57.9	62.3
12	△10 29 (64)	△2	△3	None	7.38		0 Sec	454 (1000)	10.1 (22.3)	18.46 (46.7)	31.75 (70)	8 (Cruise)	0	6.12	7.26	5.83	-	-	43.6	57.6	61.9
13	26.3 (58)	△1	△4	8 (Cruise)	6.00		40 Sec	0	11.95 (26.35)	14.36 (31.65)	23.6 (52)	0	454 (1000)	6.10	6.48	-	7.69	6.62	42.2	56.0	61.0
14	△10 23.6 (52)	△1		None	7.50		0 Sec	454 (1000)	8.93 (16.68)	14.42 (31.82)	26.3 (58)	8 (Cruise)	0	7.17	7.34	5.91	-	-	42.5	56.6	61.7
15	31 (68.5)	△2		8 (Cruise)	6.00		40 Sec	0	12.72 (28.05)	18.35 (40.45)	28.3 (62.5)	0	454 (1000)	6.11	6.49	-	7.99	6.80	46.6	61.9	67.5
16	△10 28.3 (62.5)	△2	△4	None	7.33	0 Sec	0 Sec	454 (1000)	9.77 (21.55)	18.12 (39.95)	31 (68.5)	8 (Cruise)	0	6.99	7.17	5.90	-	-	46.8	63.2	67.8

△1 LR-105 + 12 LR-101's
△2 LR-99 + 2 LR-11's
△3 Lockalloy
△4 Ablator
△5 Zero Sec Cruise Rockets
△6 Zero Sec Rocket Cruise Fuel
△7 40 Sec Rocket Cruise Fuel
△8 Max Mach Attainable at q = 47.88 kPa (1000 PSF)
△9 Vehicle has 31.75 Mg (70,000 Lb) Launch Mass Structural Capability
△10 Vehicle has structural provisions to launch with an additional cruise (8) scramjets.

Table 2 - Design Capability and Cost Summary

in the "off design" condition, equipped with cruise scramjets. The quantity of fuel was found to limit these configurations as to the maximum cruise Mach.

Cost data reflected in Table 2 are based on the "Cost Analysis" herein using the same premises as for the Phase I analysis. Included in the premises are the cost of GFAE defined in Phase I, Reference 1.

EFFECT OF VARIED DYNAMIC PRESSURE

A parametric study of the thermal protection system (TPS) was conducted of the effect of varied dynamic pressure on the total X-24C mass. This study was conducted concurrently with the analysis conducted on the THERMAL PROTECTION SYSTEM MASS INCREMENTS herein.

Analysis concluded that varying the dynamic pressure up or down from the "Baseline Point" produced little or no effect to the vehicle substructure with the major effect consisting of raising or lowering the peak temperatures to the vehicle for the parameters studies. Figure 58 shows the effect of a ±24 kPa change to the "Baseline Point" dynamic pressure of 47.9 kPa, for the parameters studies on a Lockalloy heat-sink vehicle.

Minimal mass effects were noted to the Ablator TPS vehicles due to the varied dynamic pressure, particularly since the Ablator minimum thickness beind considered are dictated by the handling limits of the TPS rather than the thermal requirement. Subsequently, lowering of the dynamic pressure has minimal effect on TPS mass.

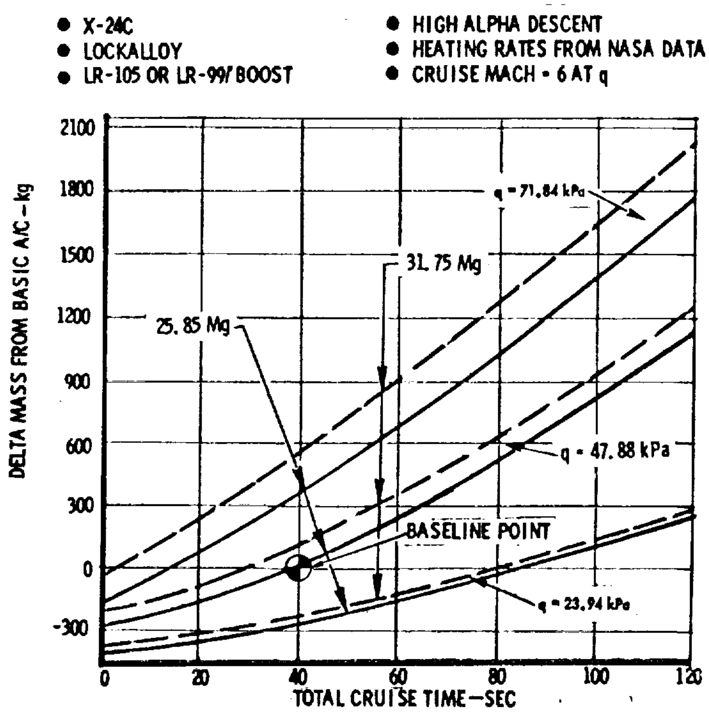


Figure 58 - Delta Mass Effect of Varied Dynamic Pressure

MASS INCREASE FOR HIGHER LOAD FACTOR

A study was conducted to determine the effect on structural mass of increasing the load factor for vehicle pullup after launch. Increased loading of the wing and fuselage shell was evaluated and increased skin thickness requirements calculated. The adjusted skin thickness requirements were then further modified to effect a linear temperature variation from the top centerline to the bottom of the fuselage. This was done to minimize the thermal stresses in the shell. The mass increment due to shell thickness increases was then calculated and an additional mass increment for necessary substructure was determined. An independent check of these calculations was made by parametric means which verified the accuracy of the mass deltas necessary to increase the initial pullup load factor. It is to be noted that the mass requirements would be somewhat less if the shell were strengthened by the addition of stiffners to the more highly stressed areas. Figure 59 depicts the results of this study for both 25.85 Mg and 31.75 Mg vehicles.

PERFORMANCE IMPROVEMENT DRAG REDUCTION

A reduction of the aerodynamic drag improves the X-24C performance in two ways. Less drag energy is expended requiring less propellant to boost to a given Mach number or boost to a higher Mach number for a given propellant. Secondarily, for the scramjet cruise design, smaller scramjets are required. These effects were evaluated and presented in Figure 60.

Assuming that the zero lift drag C_{D_O}, of the X-24C Phase II configuration can be reduced by thirty percent throughout the Mach number range of the boost, due to the wing span limitation, while mated to the B-52 launch vehicle, the drag due to lift is not changed. At the cruise condition of q = 47.9 kPa a 30 percent

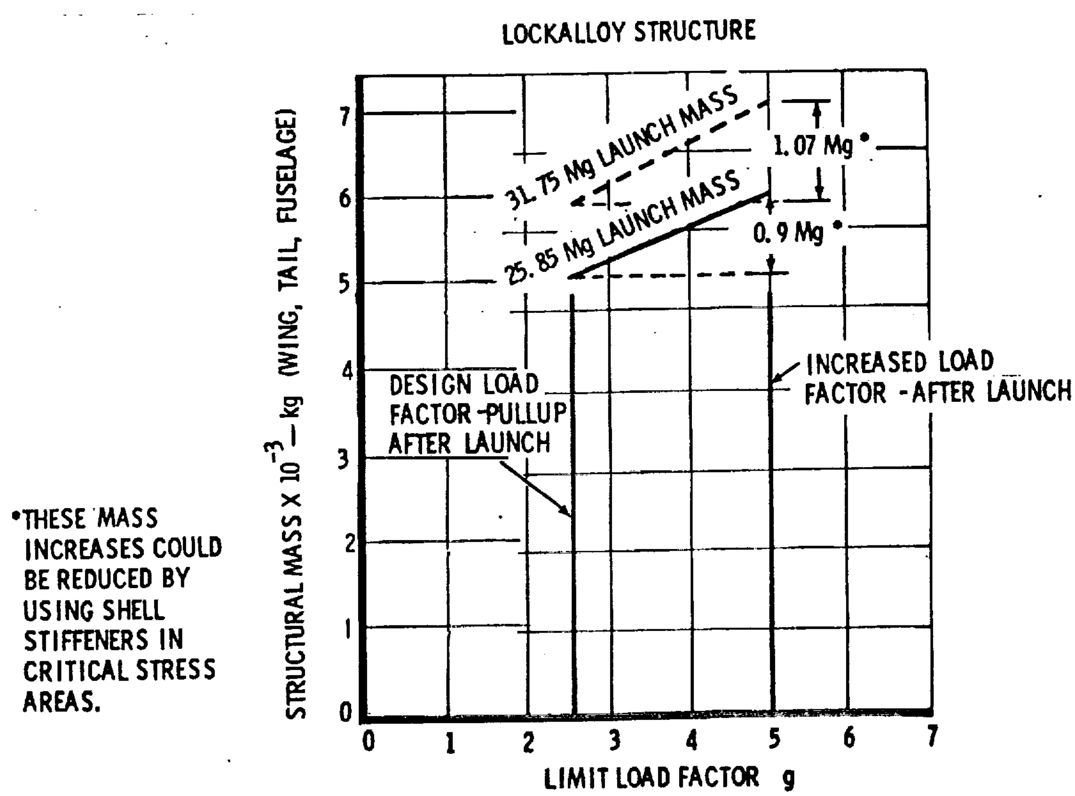


Figure 59 - Structure Mass Increase for Higher Load Factor

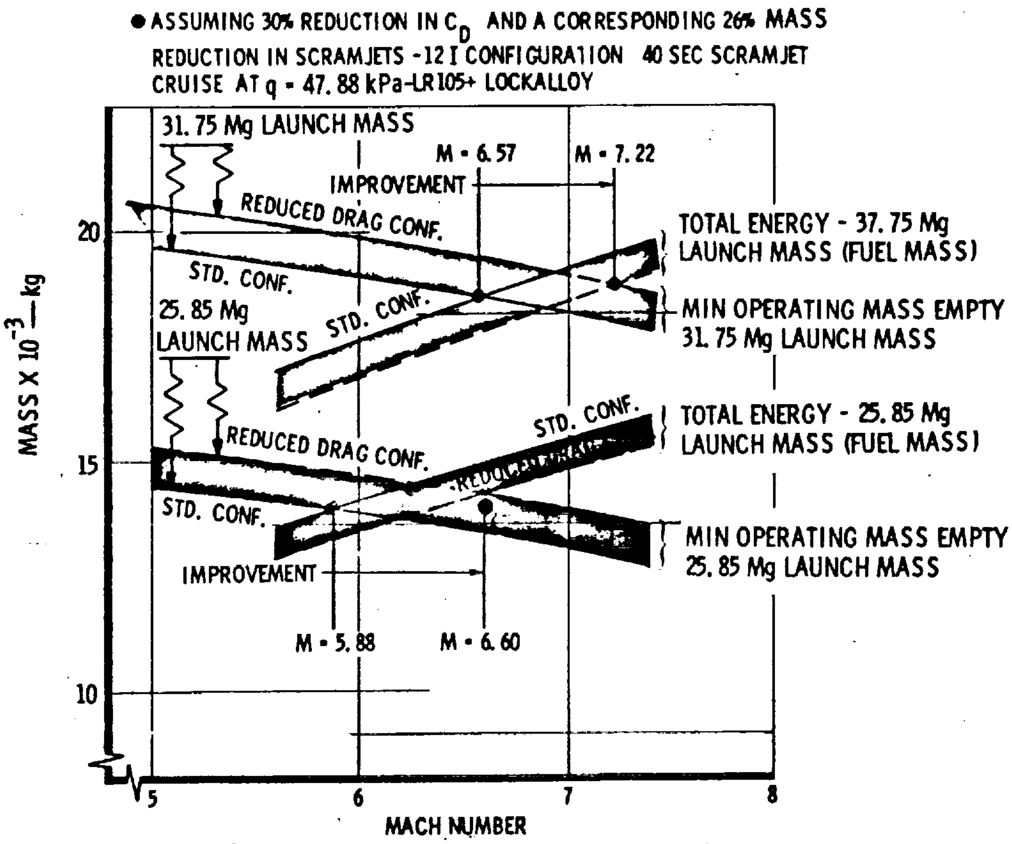


Figure 60 - Performance Improvement for Drag Reduction

in C_{D_O} will yield a 26 percent reduction in total drag. This 26 percent in total drag gives 26 percent smaller scramjets at 26 percent less mass. Figure 60 presents the results in the form of the MAXIMUM MACH NUMBER ATTAINABLE figures herein.

For the LR-105 engine, with the Lockalloy heat-sink structure, with 40 seconds of scramjet cruise at q = 47.9 kPa an increase in cruise Mach number of approximately 0.7 can be obtained from the 30 percent reduction in zero lift drag. Roughly half is obtained from propellant savings during boost and half from the scramjet mass savings. The results are shown for both 28.85 and 31.75 Mg launch mass configurations.

The maximum attainable altitude capability is evaluated for a vehicle without scramjets at a launch mass of 29.03 Mg with 18.51 Mg of rocket propellant (10.8 Mg burnout mass). This configuration can boost to a rocket fuel limited Mach number of 8.12.

The maximum altitudes are obtained by zoom maneuvers where burnout occurs during climb and the vehicle coasts to peak altitude. The maximum altitude is limited by re-entry conditions, not be available energy. Figure 61 shows the effect of 2.5 g and 5.0 g load factor limits and maximum re-entry dynamic pressure. Maximum angle of attack of 20 degrees was used during re-entry except when limited by load factor.

For the nominal vehicle with limits of 2.5 g's and q = 47.9 kPa the maximum altitude is 70.1 km. Raising the load factor limit to 5 g's would raise the maximum altitude to 85.34 km, while raising the dynamic pressure to 71.85 kPa would raise the altitude to 76.2 km.

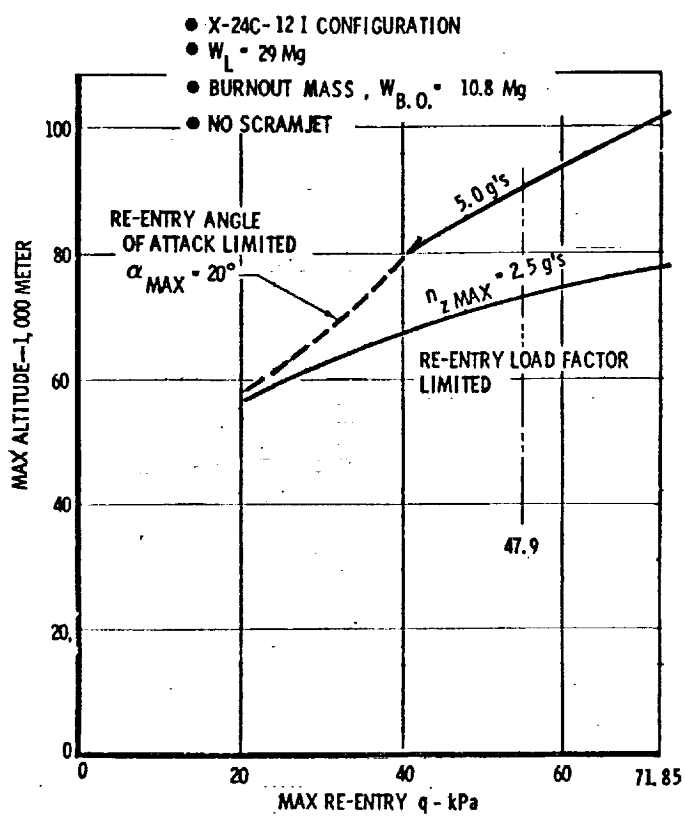


Figure 61 - Zoom Capability

INSULATION TPS PROBLEMS

Thermal plots developed for the Thermal Protection System Mass Increment analysis, herein, were the basis for establishing the percentage of vehicle surface area subject to the Reusable Surface Insulation (RIS) due to thermal gradients which exceed the Ablator TPS limit.

In addition, the Phase I Study revealed that due to fragility of the Ablator and Ablator char surface, after flight condition, a high density RSI was required as an edging member on edges subject to damage during servicing or operational usage.

Due to the added impact to the X-24C program caused by the complexity of using the RSI in conjunction with the basic Ablator TPS further analyses were conducted to uncover all related problems in this area along with its impact to the program.

Access and Service Breaks - An estimated 87.42 linear meters of service access opening and door/panel/canopy edging will require use of an RSI. Approximately 31.4 linear meters of service access and 12.6 linear meters of landing gear doors, and 37.3 linear meters of fuselage service breaks for fuel cell and engine access will utilize RSI concepts as depicted in Figure 62. Approximately 6 linear meters of canopy/sill edging will utilize an RSI installation concept depicted in Figure 62.

The 50 linear meters for service access presents an estimate based on the study to combine many of the system servicing functions to reduce the number of access ports subject to special RSI treatment. As an example of design complexity that must be avoided on the Ablator X-24C configuration is reflected in Figure 63 which depicts the total access provisions for the X-15A-2 before the Ablator series of tests. Approximately 72 of these access provisions are underlined representing "those panels thermal protection densified Ablator strips were provided around their periphery to provide a measure of protection against handling damage to the

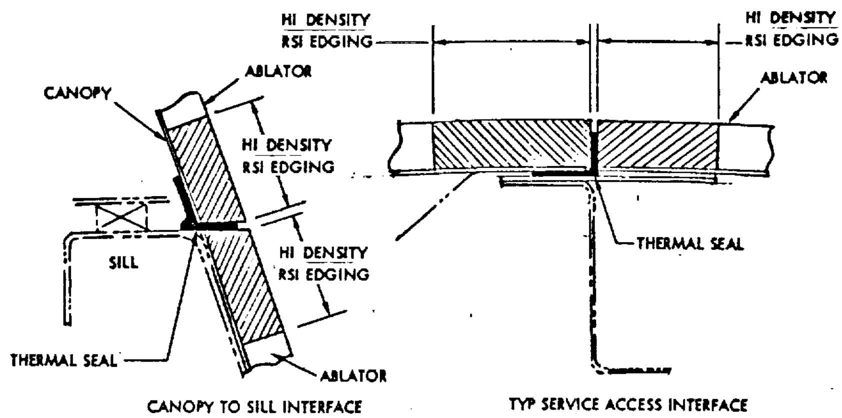


Figure 62 - Wing, Fins, Rudder Attachment and Service Breaks

Ablator edges and also contain local cutouts to provide access to fastener heads." (Page 194, Reference 4.) Combining many of these functions, as planned for the X-24C, reduces the number of separate panels.

Control Surfaces - Thermal air flow on exposed leading edges of the elevons, speed brake, and lower fuselage flap will require thermal seals, for substructure thermal control, in combination with a high density RSI. Figure 64 depicts the coneeptual design applicable to 37 linear meter of hinge surface requiring both thermal sealing and RSI edging.

Engine Shroud Provisions - Main engine and sustainer engine deflections during operation will result in impact loading along the periphery of engine nozzle and aircraft shroud. Figure 65 depicts the conceptual design of the high density RSI and thermal seal required on approximately 8.9 linear meter of interface.

Scramjet Provisions - Thermal sealing between the scramjet module and vehicle structure must be capable of slip joint action for thermal expansion and deflection (vertical) of the module overhang, outboard of the scramjet mounts. Figure 66 depicts the requirements the seal must be capable of meeting. These requirements make it obvious that an RSI in lieu of the Ablator must be used.

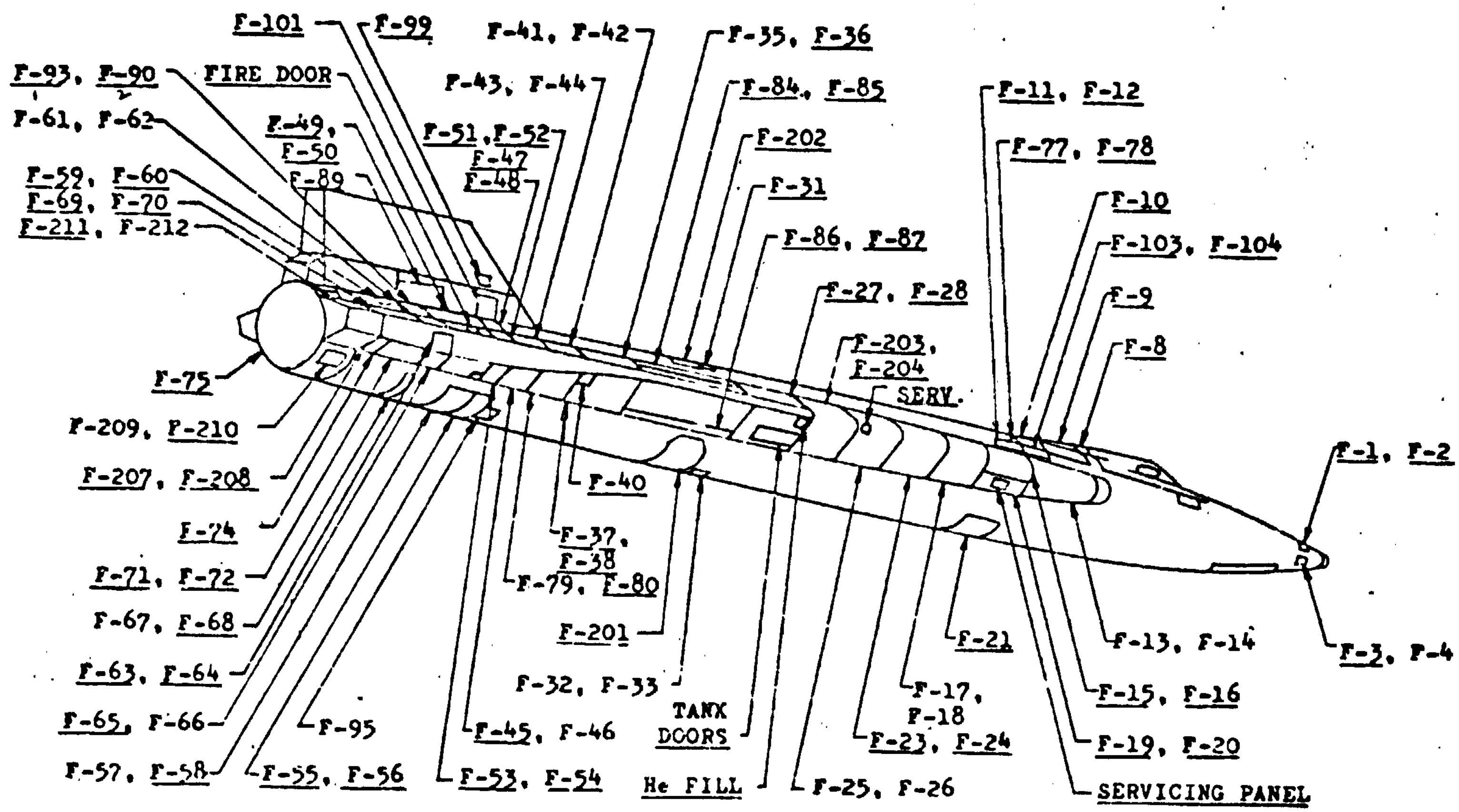


Figure 63 - Access Provisions X-15-2 Aircraft

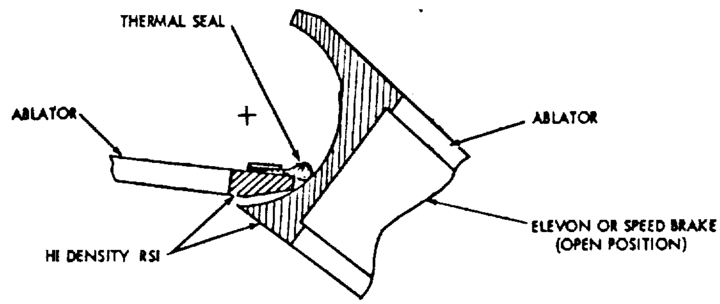


Figure 64 - Control Surfaces

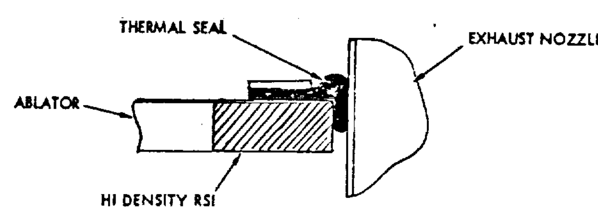


Figure 65 - Engine Shroud Provisions

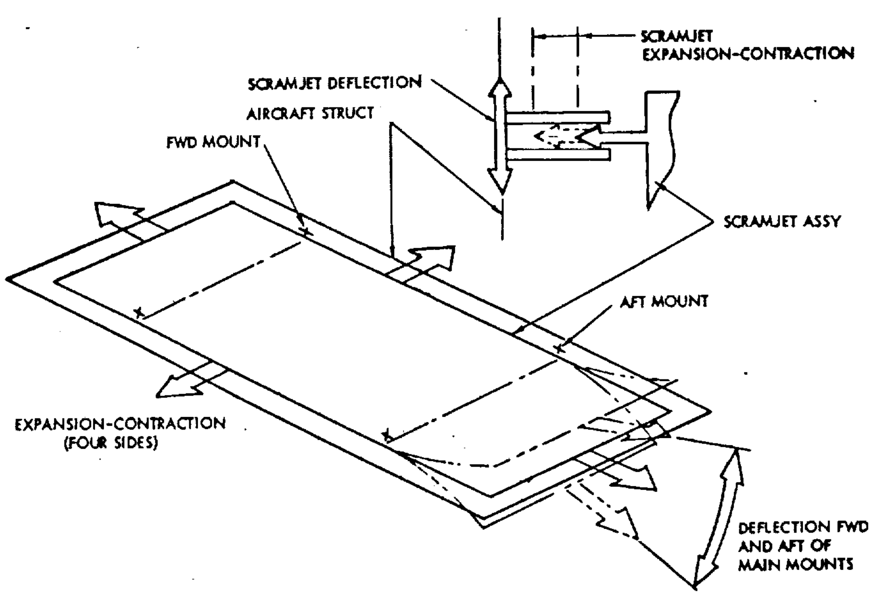
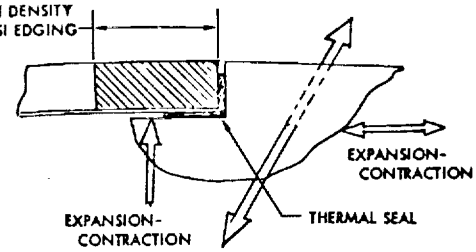


Figure 66 - Scramjet Provisions

Alternate Structure Slip Joints - The use of alternate structure assemblies i.e., wings, side fins, vertical rudder and payload bay paneling will require vehicle interfacing structure to provide both thermal sealing as well as capability of allowing for thermal expansion and contraction due to material thermal differences. Figure 67 depicts the requirements for these slip joints required at approximately 15.5 linear meter of vehicle interface where a high density RSI must be used in lieu of the Ablator.

Servicing Provisions - Under Access and Service Breaks above, RSI around service ports was discussed. Another element that must be considered pertains to the covering of certain ports after servicing, ground or post-launch, which due to their thermal characteristics do not normally require protection. These ports, i.e., for air, fuel and electrical which can be affected from proper function due to Ablator char deposits will require covers, example depicted in Figure 68, for ground installation or automatically deployed closed after vehicle launch release.

Test Instrumentation Provisions - External instrumentation sensors measuring pressure across the vehicle skin surfaceare sensitive to air flow distortions which might occur during/after charring of the Ablator surface. These sensors can also be affected by any ingestion of Ablator char deposits. To overcome the potential problem 38 mm diameter RSI buttons, Figure 69, will be provided at each sensor position. A total of 0.22 square meter of RSI material has been estimated for approximately 198 test points recommended in the USAF X-24C study, Reference 3. This number is considered conservative since in a recent NASA YF-12 Lockalloy fin program 80 pressure sensors were used on an article many times smaller than the X-24C.



NOTE: PAYLOAD BAY EXTERNAL PANEL SUBSTITUTION REQUIRES SAME PROBLEM SOLUTION; HOWEVER, FOOTAGE NOT INCLUDED IN ABOVE ANALYSIS

Figure 67 - Alternate Structure Attachment and Slip Joints

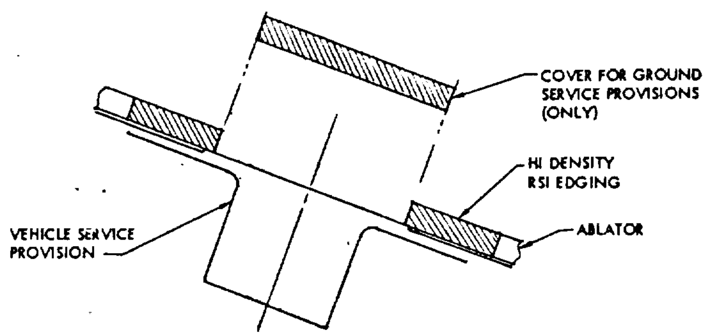


Figure 68 - Servicing Provisions

NOTE:

1. CHARRING/EROSION OF ABLATOR CAN ALSO BE DRAWN IN BY THE SENSORS INVALIDATING MEASUREMENTS AND DAMAGING INSTRUMENTATION.
2. SENSORS NOTED IN USAF AFFDL-TR-75-37 DATED MAY, 1975, YF-12 LOCKALLOY FIN REQUIRED 80 PRESSURE TEST POINTS

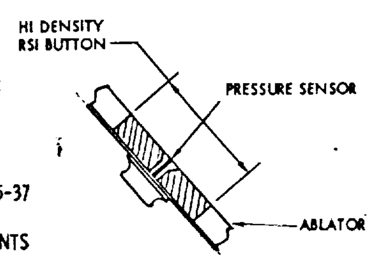


Figure 69 - Instrumentation Provisions

MAINTENANCE ASPECT

Turnaround time is defined, Reference 5, as the time from when the vehicle returns from a flight until it completes it next successful flight. This includes the time used on flight aborts and time lost due to bad weather, unscheduled maintenance, holidays, etc.

The total cost of any program is highly influenced by the calendar time a program runs. The X-24C program is tentatively scheduled to operate two vehicles for eight years and to condutt 100 flights per vehicle. This schedule will require an average of 12 flights per year per vehicle. Based on 250 working days per year, would allow a turnaround time of 20 days if the flights completed versus flights scheduled could equal 100 percent. Figure 70 summarizes the X-15 operational experience during the years 1964 and 1965, extracted from Reference 5, and shows that it was able to complete only 62 and 63 percent, respectively, of its flights. Principal reasons for the flight delays are also shown in Figure 70.

Unlike the X-15 program, the X-24C vehicle will utilize developed and validated hardware. Many of the systems will be off-the-shelf, modified for the vehicle environment, and will include build-in test equipment (BITE). Troublesome X-15 equipment, i.e., the Auxiliary Power Unit and the Ballistic Control System are not required on the X-24C. These improvements will result in a shorter turnaround time for the X-24C than with the X-15. Figures 71 and 72 compare factors affecting X-24C and X-15 turnaround times. Together with the improvement and changes expected with the X-24C program, the X-24C will be capable of successfully completing 75 percent of its planned flights. This means that to complete at least 12 flights per year, a minimum of 16 flights per year must be attempted. Using the 250 working days per year, a flight must be attempted every 15.6 days, on an average, throughout the year. In order to make this many flight attempts, it is considered necessary to be able to accomplish a mission turnaround in ten working days, Reference 1, section FLIGHT SUPPORT AND MAINTENANCE ANALYSIS.

3 SHIFTS, 6 DAY WORK WEEK, NO TPS

- FLIGHTS PER YEAR PER VEHICLE

	1964	1965
ATTEMPTED	14	17
COMPLETED	8.67	10.7
FLIGHTS COMPLETED/ATTEMPTED - %	62	63

- PRINCIPAL REASON FOR FLIGHT DELAYS - % (AFTER FLIGHT DATA HAD BEEN SELECTED)

	1964	1965
WEATHER	18	45
STRUCTURE	52	14
INERTIAL SYSTEM	4	5
STABILITY AUGMENTATION SYSTEM	5	7
ENGINE	6.5	6
MISCELLANEOUS	10.5	14
PROPELLANT SYSTEM (LESS ENGINE)	4	9
	100	100

Figure 70 - X-15-2 Operational Experience

SYSTEM	X-24C	X-15
• AUXILIARY POWER SUPPLY (APU)	BATTERIES	PROTOTYPE HI-SPEED (54,000 RPM) MECHANICAL SYSTEM REQUIRING HYDROGEN PEROXIDE (H_2O_2) PROPELLANT SYSTEM
• VEHICLE BALLASTIC CONTROL SYSTEM	NOT REQUIRED	PROTOTYPE HARDWARE - REQUIRED H_2O_2 PROPELLANT SYSTEM
• INERTIAL REFERENCE SYSTEM	MODIFIED OFF-THE-SHELF WITH BITE CAPABILITY	PROTOTYPE HARDWARE - WAS COMPLETELY REDESIGNED DURING PROGRAM
• PRIMARY FLIGHT CONTROL SYSTEM	MODIFIED OFF-THE-SHELF FLY BY WIRE WITH BITE CAPABILITY	MECHANICAL SYSTEM USING CABLES
• STABILITY AUGMENTATION SYSTEM	MODIFIED OFF-THE-SHELF WITH BITE CAPABILITY	SEMI-OPERATIONAL WITH BITE CAPABILITY (ONE VEHICLE ONLY)

Figure 71 - Factors Affecting X-24C vs X-15-2 Turnaround Time Comparisons

TASK	X-24C WITH LR-105-101 ENGINES	X-15 WITH LR-99 ENGINE
● POST FLIGHT PURGE AND INSPECTION	MUST REMOVE ENGINE	ACCOMPLISHED WITHOUT ENGINE REMOVAL
● ENGINE REMOVAL FREQUENCY	* MUST REMOVE AFTER EVERY FLIGHT TO PURGE. 16 MAN-HOURS REQUIRED TO REMOVE, PURGE, AND RE-INSTALL	WAS REMOVED EVERY 2 TO 2.5 FLIGHTS FOR REPAIRS, MAINTENANCE OR TEST RUNS. 116 MANHOURS REQUIRED TO REMOVE, TEST RUN, AND RE-INSTALL
● ENGINE RELIABILITY FACTOR	0.999	LOWEST OF ALL X-15 SYSTEMS
● PRE-FLIGHT ENGINE FIRINGS	NOT REQUIRED	REQUIRED WHENEVER THE ENGINE HAS BEEN REMOVED FROM THE AIRFRAME

* SUBJECT TO FURTHER STUDY

Figure 72 - Factors Affecting X-24C vs X-15-2 Turnaround Time Comparisons - Engine

Figure 73 depicts a scheduling of ten day turnaround activity required for a Lockalloy configured X-24C. It can be seen that for a non-heat sink configured vehicle that reserving time within this span to adequately postflight, inspect, refurbish, and preflight inspect either the Ablative or RSI TPS will be difficult to do without jeopardizing the ten day turnaround. This is especially true if other maintenance or modification activity can not be done simultaneously with TPS refurbishment. Reference 6 is a contractor prepared process manual that establishes the materials, equipment, protection and application procedures and controls that were utilized for the application and refurbishment of the Ablative TPS on the X-15A-2 aircraft. This manual shows that the elaborate contamination controls required during TPS refurbishment did not permit other vehicle maintenance activity to be conducted simultaneously. The need to protect X-24C fuel, hydraulic and personnel air systems from contaimination will be equally critical, and will require extreme care be exercised during any TPS refurbishment.

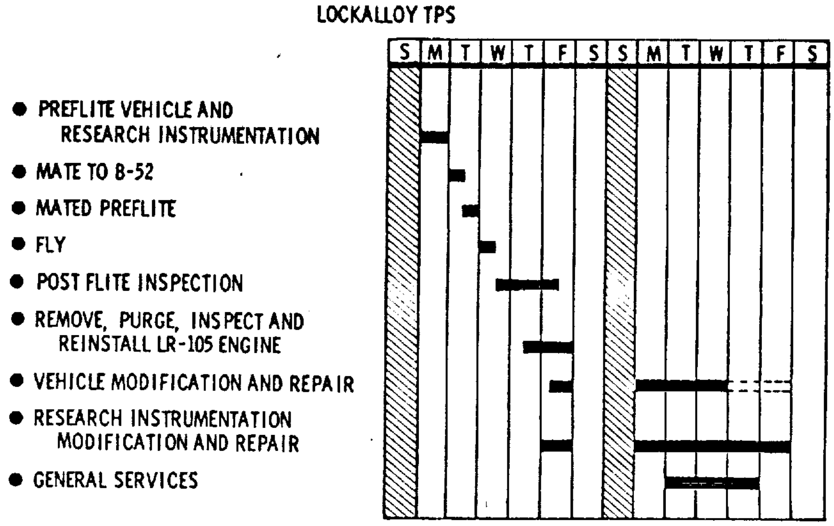


Figure 73 - Required Turnaround Schedule Capability

The Phase I study "FLIGHT SUPPORT AND MAINTENANCE ANALYSIS" section, Reference 1 estimated that an Ablative TPS vehicle would require an average 2.5 percent refurbishment per Mach 6, 40 seconds cruise flight. Assuming 185.8 square meter of X-24C vehicle wetted area and 64.55 manhours per sqpare meter to refurbish, it would require 300 manhours per flight. Be comparison, the X-15-2 after one flight (Reference 5) up to Mach 4.9 (peak velocity, no cruise) required 700 manhours and two weeks to refurbish its Ablative TPS. Complicating the task was the lox low impact sensitivity threshold of the Ablator, 27 joule, when coated with DC90-090 paint and approximately 11.5 joule for the uncoated Ablator. A current Ablative candidate for the X-24C, SLA 220, has been tested, Reference 8 and found to have a low impact sensitivity of less than 13.5 joule. Figure 74 briefly lists the effect adding Ablative TPS to the X-15-2 has on turnaround time while Figure 75 lists the additional activities found necessary to protect the air vehicle's systems during Ablative TPS refurbishment. Data presented in Figures 74 and 75 were extracted from References 5, 6 and 7,

- X-15 MINIMUM TURNAROUND TIME AS A HOT STRUCTURE VEHICLE - 6 DAYS
- X-15 AVERAGE TURNAROUND TIME AS A HOT STRUCTURE VEHICLE - 20 DAYS
- X-15 ABLATOR REFURBISHMENT REQUIRED 700 MANHOURS - 2 WEEKS
- LOX IMPACT SENSITIVITY OF ABLATOR COMPLICATED REFURBISHMENT TASK (X-15 ABLATOR IMPACT DETONATION LEVEL APPROXIMATELY 11.5 JOULE*)
- X-15 MAINTENANCE HISTORY SHOWS THAT PROPELLANT (LOX) LEAKS AND STRUCTURAL REPAIR WERE X-15 BIGGEST PROBLEMS
- WYLE LABS DATA - SLA 220 (X-24C CANDIDATE TPS) HAS LOX IMPACT DETONATION LEVEL LESS THAN 13.5 JOULE
- X-15's MA-25 ABLATOR SPRAYED WITH DC90-090 PAINT TO IMPROVE SURFACE WEAR AND RAISE LOX IMPACT DETONATION LEVEL TO 35 JOULE

*X-15 LOX IMPACT SENSITIVITY SPEC. MSFC-SPEC-106 NASA, GEORGE C. MARSHALL SPACE FLIGHT CENTER, APRIL 20, 1962 (REQUIRES 97.2 JOULE DETONATION LEVEL)

Figure 74 - Ablator Effect on X-15-2 Turnaround Time

while relating to the X-15-2 program will in part be typical for the X-24C vehicle program using the Ablative TPS.

Figure 63 is included to show the multitude of doors and access panels on the X-15-2 vehicle. The large number of doors and access panels aid in keeping maintenance hours to a minimum by providing convienent access to the vehicle's structure and system components. When a bonded TPS (Ablative or RSI) is installed on a vehicle special treatment is required initially and during the life of the vehicle around door/panel edges. This special treatment consists of installing and maintaining denser TPS material around door/panel edges, installing and maintaining effective gap seals, installing and removing fasteners hole plugs, and the extra care required in handling removable doors/panels negate to some

AT TIME THAT X-15 WENT TO ABLATOR PROTECTION SYSTEM, THE FOLLOWING ADDITIONAL ACTIVITIES WERE REQUIRED:

- ADDITION OF 14 FILTERS TO PROPELLANT LINES
- ALL OPENINGS TO AIRCRAFT INTERIOR BE SEALED USING AN ELABORATE SET OF MASKING PROCEDURES TO COMPLY WITH CLEANLINESS SPEC (MSFC-SPEC. 164) BEFORE APPLYING ABLATOR
- LOX COMPATIBLE WEAR LAYER OF DC90-090 PAINT TO BE APPLIED TO SEAL ABLATOR AND RAISE ITS LOX IMPACT ENERGY DETONATION LEVEL TO A COMPROMISED 35 JOULE (MSFC-SPEC-106 REQUIRES 72 FT-LBS)
- A DEDICATED FACILITY TO APPLY AND/OR REFURBISH THE ABLATOR
- ACCESS PANELS TO BE EDGED WITH HIGHER DENSITY ABLATOR TO PROTECT PANEL EDGES AND ATTACHMENTS
- THE ADDITION OF AN OPENABLE EYELID TO ENSURE FORWARD LANDING VISION THROUGH A CANOPY GLASS
- ALL DRAIN, VENT, AND ALIGNMENT TOOL HOLES TO BE PLUGGED PRIOR TO ABLATOR REFURBISHMENT - 78 TOTAL
- DOUBLE SEALING OR BAGGING OF 25 AIRCRAFT FEATURES, SUCH AS LOX FILL CONNECTOR, LOX VENT VALVE, BREATHING OXYGEN AND B-52 NITROGEN DISCONNECT, B-52 HOT AIR DISCONNETC, ETC., PRIOR TO ABLATOR REFURBISHMENT
- REMOVAL OF 16 AIRCRAFT SERVICE PANELS IN ORDER TO BAG OR MASK THE SYSTEM CONNECTORS WITHIN EACH COMPARTMENT PRIOR TO ABLATOR REFURBISHMENT

Figure 75 - X-15 TPS Experience

extent the maintenance manhours normally saved by the existence of a door or access panel. As noted in the Phase I Study, Reference 1, the X-24C will consider grouping of systems requiring maintenance to reduce the number of door/panels requiring special treatment.

Risks - Since program costs are the prime consideration in evaluating the candidate vehicle configurations for the X-24C program, it becomes very important to look carefully at the relative risks associated with each candidate vehicle. Figure 76 lists several hazardous conditions and asks whether the hazard can impact the materials being considered for the X-24C vehicle. Where more research or testing must be undertaken to determine a definite yes or no decision a question mark has been used to indicate this need. It is seen that for the noted considerations, Figure 76, the RSI and SLA 220 require more testing, but in answering the basic question, in the figure, Lockalloy is the only TPS for which a unanimous and confident "NO" can be given. Post flight photographs, Reference 7, of the Ablative TPS used on the X-15-2 and precautionary measures required during TPS refurbishment given in Reference 6, verify that the risk considerations, noted in Figure 76, are valid concerns to the X-24C program.

Candidate RSI and Ablative TPS's are reputed to be much more durable than the Ablative MA-25 and ESA 3560-IIA used on the X-15-2, but more testing simulating the X-24C potential flight environments, need to be run to verify improvements in durability of TPS intended for leading edges, speed brake surfaces and shock impingement areas.

Maintenance Considerations - During maintenance periods the X-24C will be exposed to all the common ground and hanger environments, including people. Figure 77 compares the TPS concepts, for the X-24C, relative to routine maintenance activities associated with almost any kind of flying vehicle. Where lack of data prevented a clear yes or no a question mark has been used on Figure 77. Since the X-24C will be required to be cleaned and maintained, it appears only the Lockalloy configured vehicle can endure cleaning, washing, or dusting. Only a Lockalloy vehicle can be safely walked on, easily inspected for structural integrity, quickly and easily cleaned of any spilled fuel or hydraulic fluids, and permit opening of LOX and/or hydraulic lines without the need of time consuming protection for Ablative or RSI covered structure.

Due to the significance of LOX impact sensitivity associated with an Ablative TPS configured X-24C it will be essential that this problem be resolved in time

COULD IT HAPPEN?	LOCKALLOY	RSI	SLA-220
• SURFACE EROSION/LOSS OF THERMAL PROTECTION IN FLIGHT	NO	YES	YES
• FIRE HAZARD/LOX IMPACT	NO	?	YES
• FIRE HAZARD/TPS CONTAMINATED BY HYDRO-CARBONS SUCH AS HYDRAULIC FLUIDS, OILS, FUELS, ETC.	NO	?	?
• INFLIGHT TPS LOSS/ALUMINUM SKIN BURN THROUGH	NO	YES	YES
• VISION OBSCURATION THROUGH WINDSHIELD/ LANDING ACCIDENT POTENTIAL	NO	NO	YES
• JEOPARDIZE TWO WEEK TURNAROUND SCHEDULE/ INCREASE PROGRAM COST	NO	YES	YES
• REQUIRE INCREASED REFURBISHMENT FOR GROWTH MACH NUMBERS 7 AND 8	NO	YES	YES

Figure 76 - Risks TPS

	LOCKALLOY	RSI	SLA-220
• CAN VEHICLE SURFACE BE CLEANED, WASHED, DUSTED?	YES	?	NO
• CAN MAINTENANCE PERSONNEL WALK ON VEHICLE?	YES	?	NO
• CAN TPS REFURB. AND VEHICLE MAINT. BE ACCOMPLISHED SIMULTANEOUSLY?	YES	NO	NO
• CAN VEHICLE STRUCTURE BE EASILY INSPECTED?	YES	NO	NO
• CAN ACCESS DOORS BE QUICKLY, EASILY REMOVED, RE-INSTALLED?	YES	?	?
• CAN TPS FLIGHT WORTHINESS BE ACCURATELY ASSESSED?	YES	NO	NO
• CAN EXTERNAL AIRCRAFT SAFETY MARKINGS BE PAINTED ON TPS?	YES	?	?
• CAN MISSION TURN AROUND ACTIVITY BE ACCOMPLISHED WITHOUT DAMAGING TPS? (ONCE, TWICE, --- 99 TIMES)	YES	?	?

Figure 77 - Maintenance Considerations

to support the development and design of the X-24C. Figure 78 delineates the tests and design areas which must be taken into account to assure LOX compatibility and vehicle safety.

Risks TPS - One of the prime considerations in evaluating the candidate vehicle configuration for the X-24C program relates to the risk involved in the selection of the type of external surface used for the X-24C. Figure 79 summarizes the assessment of the TPS risks as defined in the Phase I Study report, Reference 1. This assessment was made for a vehicle capable of achieving Mach 6 or less. In assessing the "off design" potentials in this study the increase in Mach number, Figures 80 and 81, has no apparent assessment effect on configurations using Lockalloy or LI-900 RSI. But, a noticeable impact does occur on the Ablative TPS. This is primarily caused by the increase in RSI in lieu of Ablative TPS, reference "THERMAL PROTECTION SYSTEM MASS INCREMENTS" herein, which increases initial development cost, refurbishment cost, as well as increasing the difficulty in assessing proof of operational integrity. Where more testing is necessary to determine a definite yes or no a question mark has been used in the three assessment figures.

Propulsion System - Phase I of this study included an assessment of the two candidate propulsion systems being considered for the X-24C program. Phase I recommendations conclude the LR-105 with 12 LR-101 engines to be the recommended choice with the LR-99 with two LR-11 engines as potential backup. During the Phase II effort it has been necessary to evaluate the propulsion systems during evaluation of vehicle growth and "off design" potential. The recommendation made at the conclusion of Phase I remained unaffected by this study. Figure 82 summarizes the concern defined as part of the Phase I study relating to the LR-105 engine. Investigations with the source vendor during this study did not produce any changes. Figure 83 summarizes concerns relating to the use of the LR-99 engine in the X-24C program. The findings have been expanded based on data extracted from Reference 5 and 7, relating to the use of this engine on the X-15-2, and which are considered applicable to the X-24C program. Figure 83

TEST DESIGN, SPECIAL EQUIPMENT REQUIREMENTS

- TESTS
  - DETERMINE LENGTH OF TIME ABLATIVE TPS IS SENSITIVE TO IMPACT OR COMBUSTION AFTER LOX SPILL
  - DETERMINE FEASIBILITY OF APPLYING A LOX COMPATIBLE SEALANT OVER TPS BETWEEN FLIGHTS
  - DETERMINE AMOUNT OF WETTING OF AFT FUSELAGE BY FUEL AND LOX DURING FLIGHT FROM PRIME PURGE AND JETTISON
- SPECIAL EQUIPMENT
  - SHIELDS, COVERS
  - SNIFFERS
  - DOUBLE SEALS, FUEL/LOX VENTS, ETC.
- DESIGN
  - LOCATION OF PROPELLANT FILL PORTS
  - LINE LEAKAGE AND VENTILATION
  - ACCESS DOORS TO VITAL SYSTEMS
  - JACK POINTS

Figure 78 - Significance of LOX Impact Sensitivity

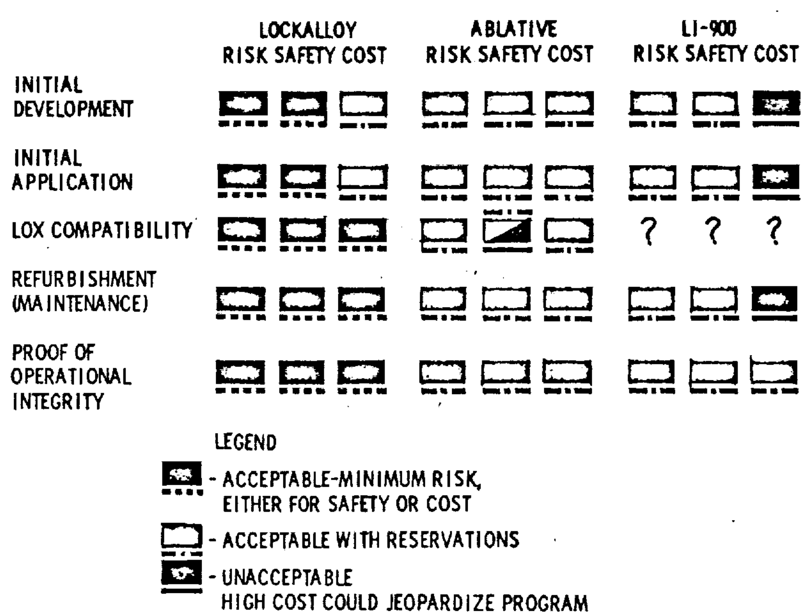


Figure 79 - Summary Comparison Risk Assessment of TPS @ M = 6

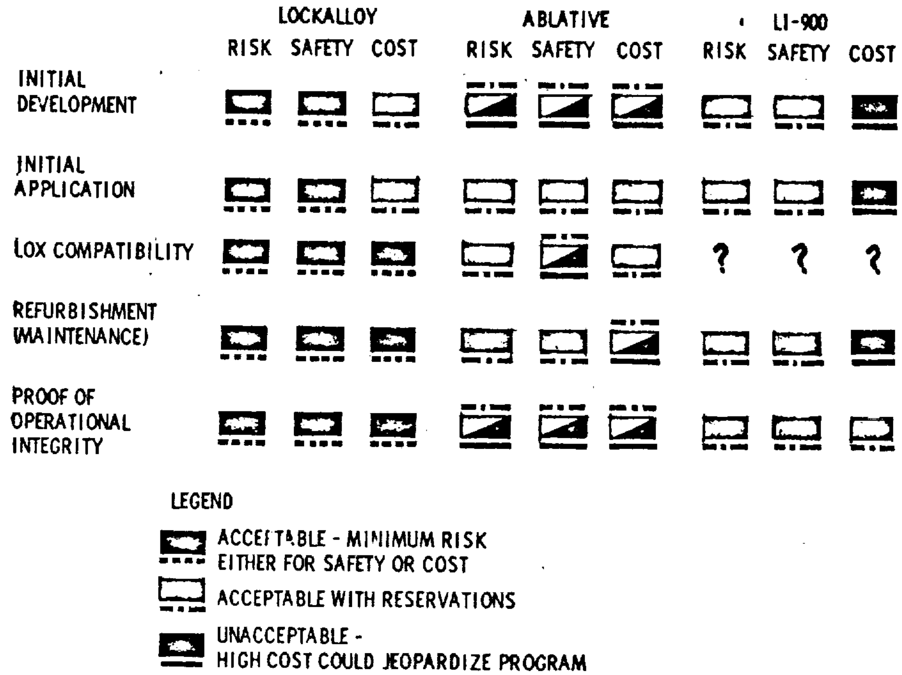


Figure 80 - Summary Comparison Risk Assessment of TPS @ M = 7

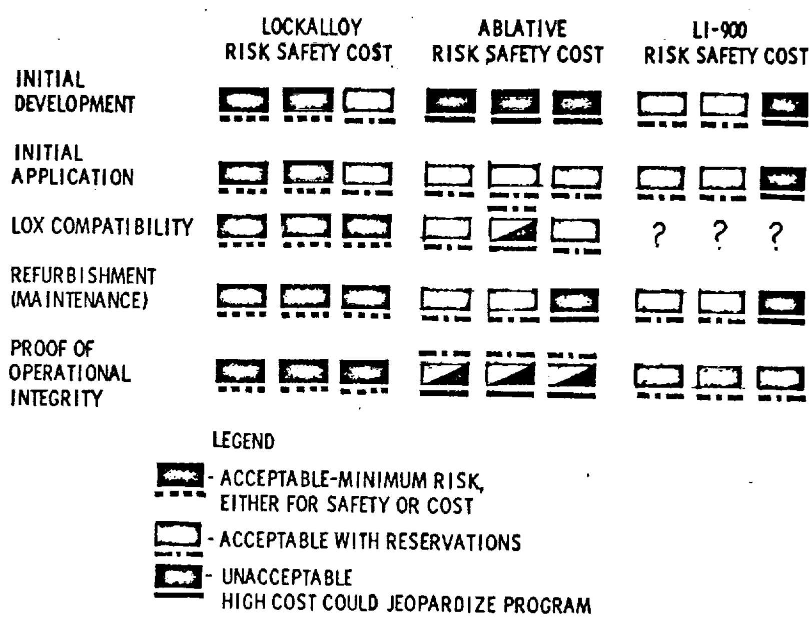


Figure 81 - Summary Comparison Risk Assessment of TPS @ M = 8

THE FOLLOWING ARE UNDER STUDY BY ROCKETDYNE UNDER CONTRACT WITH USAF-RPL

- SHUTDOWN AND PURGE STILL ALLOWS SOME FUEL AND LOX TO COLLECT IN COMBUSTION CHAMBER WITH THE POSSIBILITY OF EXPLOSION OR DAMAGE ON LATER CONTACT WITH LOX
- ROCKETDYNE RECOMMENDS THE LR-105 BE REMOVED AFTER EVERY FLIGHT FOR FLUSHING AND PURGING IN A VERTICAL ATTITUDE
- REQUIRES ADDITIONAL RELIABILITY TESTING FOR HORIZONTAL OPERATION
- NUMBER OF X-24C FLIGHTS ACHIEVABLE PER ENGINE STILL TO BE DETERMINED
- AVAILABILITY TO X-24C PROGRAM

Figure 82 - Concern about LR-105 Usage

also reflects Lockheed ADP judgement factors. Figure 84 lists considerations relating to the two sustainer engine LR-101 and LR-11 noted from discussions, with the source vendor, which are considered applicable to the X-24C program.

- THE NUMBER OF PLUMBING DISCREPANCIES (LEAKS) WRITTEN AGAINST THE ENGINE WAS EXCEEDED ONLY BY THE PROPELLANT - PNEUMATIC SYSTEM DISCREPANCIES
- LR-99 ENGINE SYSTEM WAS A MAJOR FACTOR IN X-15 PROGRAM DELAYS
- AVERAGE 2 TO 2.5 FLIGHTS FOR ENGINE REMOVAL
- THE ENGINE TURBOPUMP HAD THE HIGHEST COMPONENT FAILURE RATE - .855 FAILURES PER FLIGHT. THE NEXT HIGHEST RATE WAS .377 FAILURES PER FLIGHT
- SPARES AVAILABILITY - SHELF LIFE OF ALL SOFTWARE (O-RINGS, SEALS, ETC.) HAS BEEN EXCEEDED. REPLACEMENT AND/OR REQUALIFICATION IS REQUIRED
- SPARE TURBO PUMPS DO NOT EXIST
- THE SPARE SUPPLY SYSTEM REQUIRES REACTIVATION
- A TECHNICAL SUPPORT TEAM REQUIRES REVITALIZING
- THE LR-99 ENGINE NEVER HAD A SATISFACTORY PRODUCT IMPROVEMENT PROGRAM AND NOW EXISTS AS SEVERAL LAYERS OF ENGINEERING CHANGE ORDERS UPON THE ORIGINAL DESIGN.
- REQUIRES A DEDICATED PROPELLANT SYSTEM TO DRIVE THE PRIMARY PROPELLANT TURBO PUMP
- REQUIRES THE DESIGN, FABRICATION, AND TESTING OF THE NOZZLE EXTENSION
- PERFORMANCE - THRUST AND SPECIFICS ARE LESS THAN OBTAINABLE WITH THE LR105

Figure 83 - Concern about LR-99 Usage Based on X-15-2 Experience and other Judgment Factors

	LR-101	LR-11
• RELIABILITY	0.9994	8 PRESSURE SENSING IGNITERS PER VEHICLE SENSITIVE TO COLD SOAKING
• AVAILABILITY - COMPLETE ENGINE	IN PRODUCTION 15-20 IN STORAGE	OUT OF PRODUCTION 7 COMPLETE ENGINES
• AVAILABILITY - SPARE COMPONENTS	READILY AVAILABLE DUE TO PRODUCTION STATUS	16-20 SPARE COMBUSTION CHAMBERS SEVERAL (?) TURBO-PUMPS
• COMPATIBILITY WITH BOOSTER FUEL	USES SAME FUEL AS LR-105	TESTING REQUIRED TO CONFIRM COMPATIBILITY WITH AMMONIA USED WITH LR-99
• MAINTENANCE COMPLEXITY	ROCKETDYNE SUPPORT REQUIRED REQUIRED ONLY FOR MAJOR OVERHAUL	THIOKOL SUPPORT REQUIRED FULL TIME
• AGE	AVAILABLE	AVAILABLE

Figure 84 - Sustainer Engine Considerations

COST ANALYSIS

For Phase II of the configuration study the same premises are used for cost estimating as for Phase I with one exception. That exception is the result of a re-evaluation of the ablator type TPS. Phase II engineering estimates indicate that the use of ablator type material should be limited to surface areas on the vehicle with temperatures lower than 922 K rather than the 1033 K assumed for Phase I. This increases the requirement for higher density RSI type TPS materials. For example, a vehicle designed for Mach 6, 40-second cruise requires:

	Phase I Estimate	Phase II Revised
Surface area:		
Ablator	80%	50%
RSI type	20%	50%
Average cost per m^2 (based on two vehicles)	\$17,018	\$25,694

This is further illustrated in the THERMAL PROTECTION SYSTEM MASS INCREMENTS section herein.

In addition, in-house testing by the airframe manufacturer has been increased to provide for the use of TSI tile of higher density and q factor than is currently developed for the Space Shuttle. However, the cost of full scale RSI material development by sources such as NASA-Ames or Lockheed Missiles and Space Company that may be required is not included.

Based on an engineering evaluation, the complexity factors and installation costs of the LR-99 plus two LR-11 rocket engine combinations are estimated to be equal to those for the LR-105 plus two LR-11 propulsion systems from Phase I.

Other costing premises that remain unchanged from Phase I are:

- Costs are estimated in January 1976 dollars
- Excludes aero configuration development wind tunnel testing which is separately estimated at an additional $1.5 mission
- Excludes B-52 carrier modification
- Excludes flight test and support after delivery
- Excludes the cost of propulsion systems (rocket engines)
- Excludes the cost of flight test instrumentation and payload/experiment development

Table 2 lists the capabilities and cost of sixteen selected configurations. For these varied configurations, costs are developed which consider the airframe (DCPR) manufactured mass, vehicle surface area, type of TPS and rocket propulsion systems for each. A further breakdown by major cost element for vehicle item numbers 1, 3, 5 and 7 is provided as follows:

(January 1976 dollars in millions)

	#1 Base-line	#1 Pro-pulsion	#1 TPS	#1 Total	#3 Base-line	#3 Pro-pulsion	#3 TPS	#3 Total
One Vehicle:								
Engineering	$12.2	$ 3.2	Included in Baseline	$15.4	$11.6	$ 3.5	Included in Baseline	$15.1
Tooling	8.2	3.0		11.2	7.6	3.4		11.0
Mfg. Labor	7.9	3.0		10.9	7.4	3.3		10.7
Mfg. Mat'l. & Equip.	5.8	1.1		6.9	5.7	1.1		6.8
GFAE	.3	0		.3	.3	0		.3
Totals	$34.4	$ 10.3		$44.7	$32.6	$ 11.3		$43.9

	Base-line	Pro-pulsion	TPS	Total	Base-line	Pro-pulsion	TPS	Total
Two Vehicles:								
Engineering	$12.5	$ 3.3	Included in Baseline	$15.8	$11.8	$ 3.7	Included in Baseline	$15.5
Tooling	8.4	3.2		11.6	7.8	3.6		11.4
Mfg. Labor	14.1	5.4		19.5	13.2	5.9		19.1
Mfg. Mat'l. & Equip.	9.6	1.8		11.4	9.4	1.8		11.2
GFAE	.7	0		.7	.7	0		.7
Totals	$45.3	$13.7		$59.0	$42.9	$15.0		$57.9

TPS: Lockalloy (#1) — Lockalloy (#3)

Engines: LR-105 + 12 LR-101 (#1) — LR-99 + 2 LR-11 (#3)

(January 1976 dollars in millions)

	#5				#7			
	Base-line	Pro-pulsion	TPS	Total	Base-line	Pro-pulsion	TPS	Total
One Vehicle:								
Engineering	$10.0	$ 3.1	$3.4	$16.5	$ 9.8	$ 3.4	$3.2	$16.4
Tooling	8.3	3.0	.8	12.1	8.1	3.3	.6	12.0
Mfg. Labor	8.7	3.0	.7	12.4	8.3	3.4	.6	12.3
Mfg. Mat'l. & Equip.	2.8	.6	3.3	6.7	2.8	.6	2.7	6.1
GFAE	.3	θ	θ	.3	.3	θ	θ	.3
Totals	$30.1	$ 9.7	$8.2	$48.0	$29.3	$10.7	$7.1	$47.1

	Base-line	Pro-pulsion	TPS	Total	Base-line	Pro-pulsion	TPS	Total
Two Vehicles:								
Engineering	$10.3	$ 3.3	$ 3.8	$17.4	$10.1	$ 3.6	$ 3.5	$17.2
Tooling	8.8	3.1	.8	12.7	8.6	3.4	.6	12.6
Mfg. Labor	15.4	5.6	1.3	22.3	14.9	6.2	1.1	22.2
Mfg. Mat'l. & Equip.	3.6	.9	6.2	10.7	3.5	1.0	5.3	9.8
GFAE	.7	θ	θ	.7	.7	θ	θ	.7
Totals	$38.8	$12.9	$12.1	$63.8	$37.8	$14.2	$10.5	$62.5
TPS:	Ablator					Ablator		
Engines;	LR-105 + 12 LR-101					LR-99 + LR-11		

These breakdowns are typical of the sixteen configurations listed.

From the table of selected configurations an envelope of vehicle cost vs. launch mass can be derived. Based on the X-24C configuration designed to cruise scramjets for 40 seconds at Mach 6.0, the alternative TPS and propulsion systems provide upper and lower cost limits as shown on the Figure 85. All vehicles shown on this chart are designed for the same mission and the only variables are launch mass and related cost. The upper range of launch mass at 31.75 Mg is constrained by the B-52 limit. The other plot points are established as the cost of the minimum vehicle capable of meeting the performance parameters with the respective TPS and propulsion systems.

Vehicle cost vs. Mach number can also be derived from these data. This is displayed on Figure 86. The full range of performance potential is shown. This chart also shows a region of anticipated improvement in performance and cost with the drag reductions, anticipated from the drag improvement analysis herein, that may be achieved during configuration refinement in Phase III of the study.

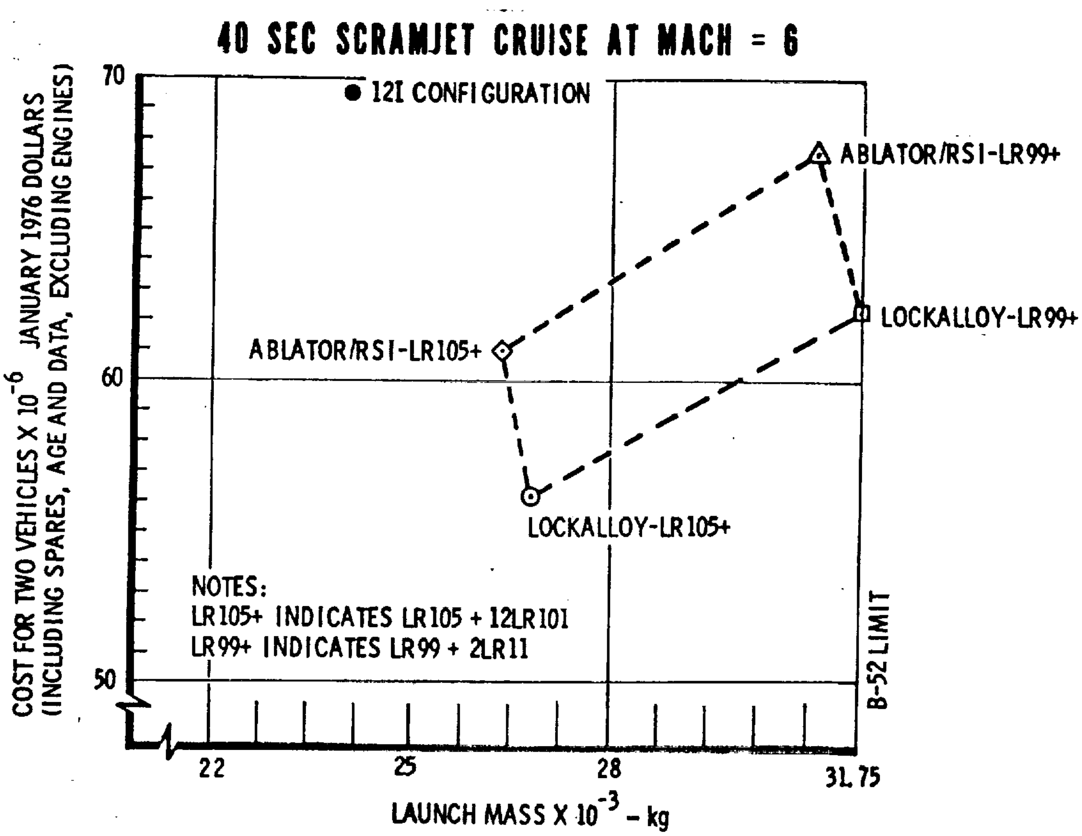


Figure 85 - Envelope of Cost vs Launch Mass

The X-24C vehicle with Lockalloy skin as a combined structure and heat sink as TPS and the LR-105 rocket engine with twelve LR-101 engines for cruise offers the best performance vs cost as shown by this study.

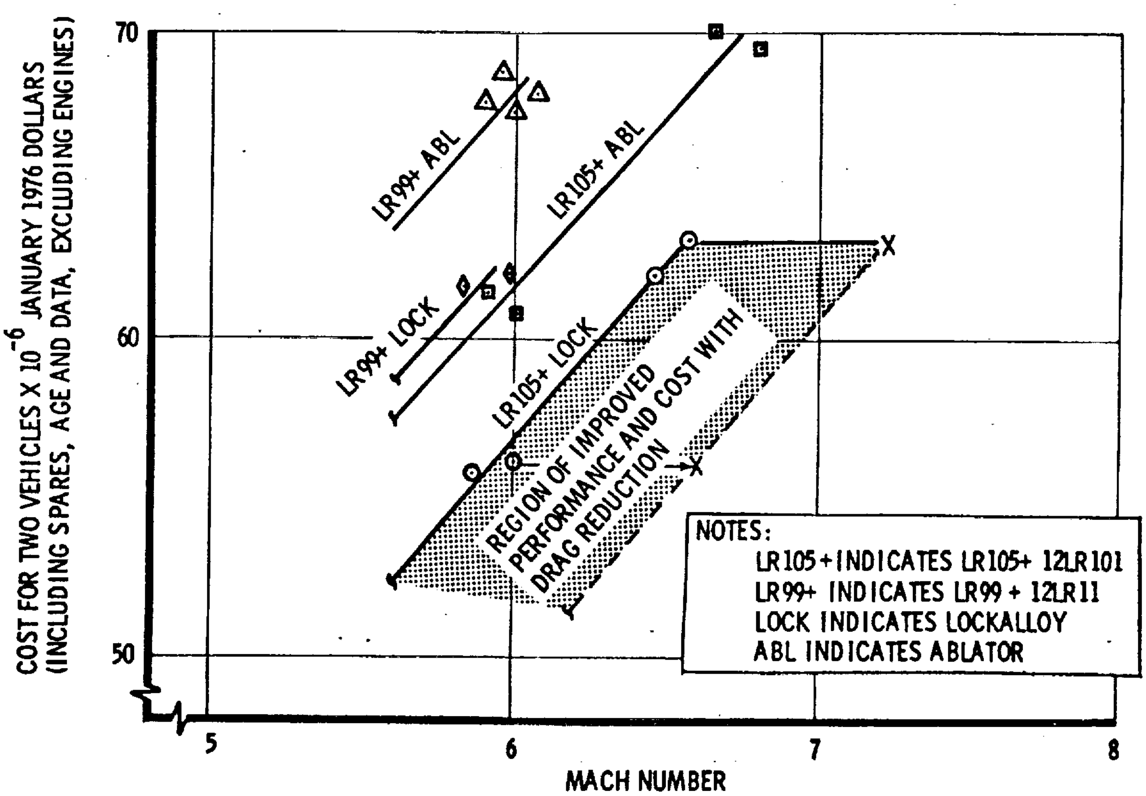


Figure 86 - Cost vs Mach No with 40 Second Scramjet Cruise

CONCLUSION AND RECOMMENDATIONS

A study of the impact of the performance growth potentials on the size, mass, cost and utility of the X-24C research vehicle configurations, which emerged from the conclusions of the Phase I study program, has been performed. Experience and results from the X-15-2 program was drawn upon to support similarities to the X-24C program.

Recommendations reached by the study include the following:

(1) SLA 220 and LI-900 or other available RSI should be abandoned and a Lockalloy heat sink configuration selected because it insures the following advantages:

- Greatest flight safety
- Least fire hazard - inflight or ground
- Fastest mission turnaround
- Least refurbishment cost per flight
- Simplest, most reliable solution to the airframe thermal protection problem
- Simplest solution to the problem of thermal seals at all service joints
- Does not release particles that deposit on canopy glass, service connections, sensors or which can ingest in scramjet engines
- Cleanest aerodynamic surface
- Greatest growth potential for increased flight Mach numbers.

(2) Selected one final concept for the Phase III configuration development that will provide the best attainable X-24C performance at the lowest cost. This concept provides:

- 31.75 Mg launch mass (B-52 limit)
- LR-105 plus 12 LR-101's for the primary propulsion system
- Lockalloy for the combined structure and thermal protection system (TPS)

(3) Other candidate vehicle concepts, including TPS and propulsion, are not ruled out for X-24C procurement. They are within the feasibility envelope as established by the Phase III selection concept.

PREVIEW OF THE PHASE III STUDY

Phase III will evolve a candidate vehicle configuration which takes into account simultaneously the results of the Phase I and II studies, and all available hypersonic technology.

REFERENCES

1. Combs, H.G., "Configuration Development Study of the X-24C Hypersonic Research Airplane - Phase I," NASA-CR-145032, dated December 1976

2. Brackeen, R.E. and Marcy, W.L., "X-24B Growth Version Feasibility Study," Technical Report AFFDL-TR-73-116, October 1973.

3. Experiments Impact on X-24C, AFFDL-TR-75-37, dated May 1975.

4. Price, A.B., "Design Report Thermal Protection System X-15A-2, NASA-CR-82003, January 1968.

5. Love, J.E. and Young, R.; "Survey of Operation and Cost Experience of the X-15 Airplane as a Reusable Space Vehicle System," NASA TN-D-3732, November 1966.

6. Thompson, J.T. and Price, A.B.; "Application and Refurbishment Procedures X-15-2 Thermal Protection System," NASA CR-96000.

7. Armstrong, J.G.p "Flight Planning and Conduct of the X-15-2 Envelope Expansion Program," FTC-TD-69-4, July 1969.

8. Wyle Laboratory; "SLA 220 LOX Impact Sensitivity Test Results," Report No. 57.008, dated March 1976.