

"PEGASUS" - A DESIGN CONCEPT FOR A V.I.P. ORBITAL/GLOBAL ROCKET TRANSPORT

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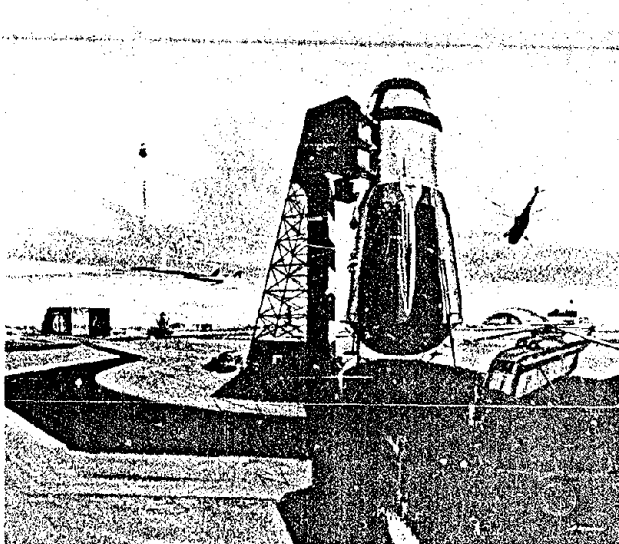
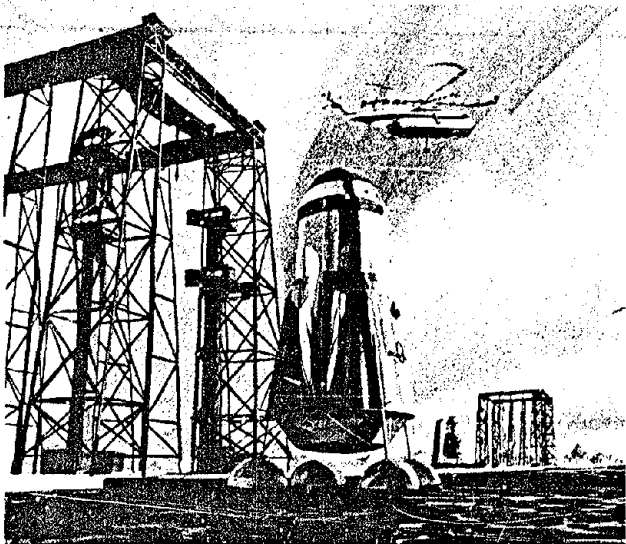
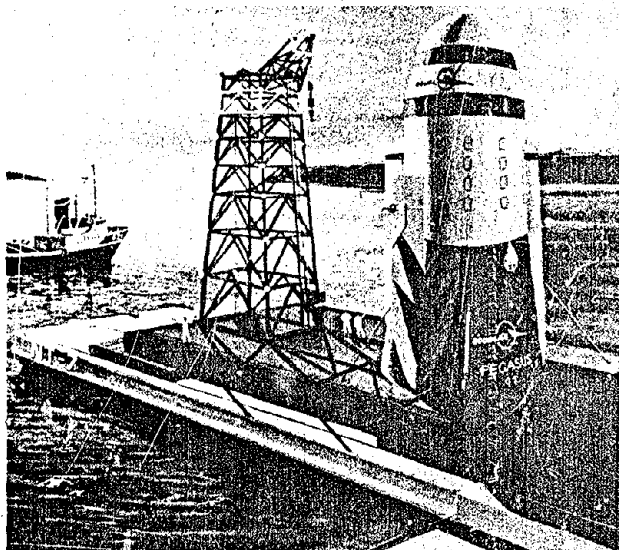
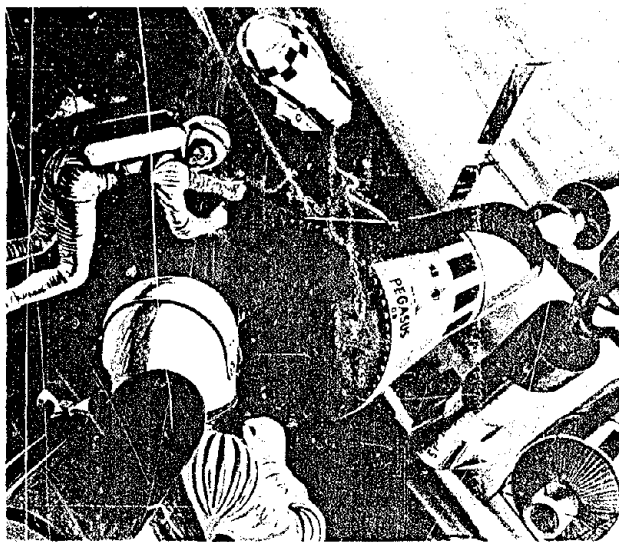
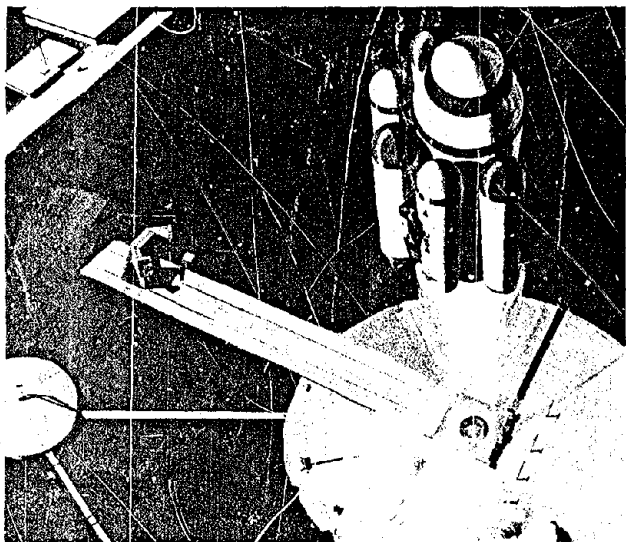
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ABSTRACT

Just as the Saturn class of space launch vehicles is a logical progression from the expendable ICBM of yesterday, a new class of carrier for high-income passengers and high-priority cargo may evolve from the reusable booster of tomorrow. This paper describes the results of a study by the Douglas Aircraft Company, Inc. to determine operational and economic aspects of such a potentially feasible vehicle. Evolution of the hypothetical transport system, termed Pegasus, is based on the premise that after vehicle survival of atmospheric penetration has become commonplace, a reusable booster of the Saturn V payload class (100 tons to orbit) may be developed and later adapted for ballistic surface-to-surface payload delivery.

Because these boosters will be used repeatedly, the extensive preflight preparations and exhaustive checkout procedures which are now mandatory every time an expendable booster is launched may eventually be reduced to a level comparable to that of commercial airliners. Only when such simplified launch operations can be realized would the Pegasus hypothetical vehicle prove to be operationally feasible. A vertical takeoff and landing (VTOL) rocket, with a Saturn V orbital capability could be adapted to deliver 170 passengers and 18 tons of cargo to any point on Earth within 45 min. at average speeds of 17,000 mph, without subjecting the passengers to any more than 3 g's during boost or atmospheric entry.

Such a transport could ballistically rocket first-aid personnel and emergency equipment to any disaster area on Earth in a fraction of the time required by the proposed supersonic transport. After propulsive terminal velocity cancellation, vertical touchdown of Pegasus could be accomplished at a foreign spaceport on four extensible legs, in a manner similar to the Apollo method of landing on the moon. Without doubt, Pegasus could be the most revolutionary advance in commercial transportation since the airplane.

This global transport vehicle, utilizing many newly acquired techniques of space flight, could be operational by the early 1980's. The vehicle is readily adaptable to satisfy the commercial market which may develop in the next two decades for a passenger transport for round trips to Earth-orbiting space stations. This type of rocket-powered hypersonic transport is an extremely promising means of realizing man's long-sought flight capability outside the Earth's atmosphere, where prolonged aerodynamic heating is nonexistent. By the 1980's, the probable state of international relations may well necessitate immediate diplomatic contact. Time is one of our most precious possessions and, undoubtedly, more speed (than the supersonic transport can provide) will be required by heads of state, executives, doctors, patients, professionals, and others in the 1980 era.

INTRODUCTION

Today, the engineering community looks forward with confidence to man's first flight to the moon. Equally thrilling will be a 2,000-mph flight in the supersonic transport (SST). A brief period of technical deliberation on these two seemingly unrelated activities would suggest that perhaps the basic technology needed for one objective could also satisfy the requirements for the other. Clearly, if passengers can be rocketed the 240,000 statute miles to the moon, they can be rocketed much more easily and safely for only 12,400 statute miles, or halfway around the globe. Moreover, travel at suborbital speeds of 17,000 mph, instead of the 2,000 mph of the SST, offers impressive reductions in transit time. Thus, while we investigate the use of air-breathing machines propelled at speeds for which they are perhaps not well adapted, we cannot afford to dismiss the rocket transport as a far-fetched impractical pipe dream. We must design today as if the next 10 years had already passed.

The SST, operating within the atmosphere at all times, would be navigating within the "thermal thicket" for all of its high-speed flight. That is, its structure must withstand long-sustained heating without excessive loss of strength. The sonic boom problem, and its effect on populated areas, must be solved to the satisfaction of all nations along each route. In addition, the SST requires longer runways at existing airports, extremely precise traffic control, and increasingly complex landing aids.

By comparison, the VTOL rocket transport, or "domesticated missile," conceivably could incorporate propulsion, guidance, and structural components (or maneuver and landing techniques) which have been developed under the far more severe environments of the Apollo program. It should be pointed out that although the conceptual vehicle defined in this paper has been subjected to only a cursory analysis, its incorporation of advanced propulsion technology would necessitate a radical departure from the conventional design approach of today's expendable boosters. Nevertheless, it would not be premature to speculate on how such an advanced aerospace transportation system might operate in order to identify the problem areas and systems criteria for long-range planning and to establish priority for supporting research. Such speculation was made in a recently concluded study by the Douglas Aircraft Company, Inc.

Although the relatively shallow cost analysis conducted to date indicates that a liquid oxygen/

liquid hydrogen propellant system cannot compete economically with kerosene-burning airplanes in the Earth surface transportation market (from one point on the Earth to another), economy or passenger comfort alone do not necessarily dictate the desirability of a system. Reduced transit time, the only advantage offered by the SST, is also a predominant factor. The transportation system concept described in this paper was evolved as a potential means of providing the ultimate speed for travel on the Earth's surface (a velocity greater than this would propel the travellers into orbit).

The value of a high-capacity aerospace workhorse was demonstrated during the international crisis which precipitated the Berlin airlift of April 1948 to September 1949. During this 18-month period, 2.3 million tons of food and coal were airlifted by British and American planes into West Berlin. The need for fast reaction time as well as high capacity is re-emphasized during catastrophes such as the Alaskan earthquake of 27 March 1964. This disaster devastated a large portion of the runway at Anchorage International Airport, the nerve center of Alaskan commercial aviation. Communications were in ruins. Tons of water, food, medical supplies, and field hospitals had to be immediately airlifted to Elmendorf Air Force Base and then distributed to the badly stricken areas south of Anchorage. During the first three days of the emergency, military planes landed more than 300 tons of supplies in Alaska. The largest single item was an 800-kw, 34,000-lb. gas turbine generator. Other heavy cargo included nine 25,000-lb. water purifiers, each of which could process 1,500 gal. of water per hour. In slightly more than a week, a single commercial carrier rushed nearly 200 tons of cargo into the stricken zone.

This was truly an amazing performance for present-day transport vehicles. However, our Pegasus vehicle, as presently envisioned, would be capable of delivering 170 first-aid personnel and 18 tons of emergency equipment from the Pacific Missile Range to Anchorage in just 17 min. despite the condition of the runway at the landing field. For this same 2,000-n.mi. flight, the SST would require 1.2 hours. Existing turbojet aircraft, on the other hand, required 3 hours to travel that portion of the flight just from Seattle, Washington to Elmendorf AFB. Thus, the speed and resulting tonnage possible with the Pegasus would open new vistas in disaster relief (including the reduction or elimination of destructive aftereffects).

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PROJECTION FROM HISTORICAL TRENDS

Figure 1 presents a cursory evaluation of historical trends of Earth transport (and orbital transport) systems, with an extrapolation to the near future. By 1958, when the jet liner became operational, it is estimated that a total of one million tons of commercial payload (excluding the military cargo of the Berlin airlift) had been delivered by air carriers to global distances of 2,500 mi. or more. An extrapolation of the upper curve suggests a possible justification for the development of a rocket transport system by the early 1980's.

The lower curve demonstrates that during the past five years both the number of space vehicles launched and the size of the payloads they carry have grown steadily. The accomplishments to date as well as the plans for the future suggest that activity in near space (and in deep space) will continue to grow to the extent that relatively routine space operations and launchings should occur in the not-too-distant future.

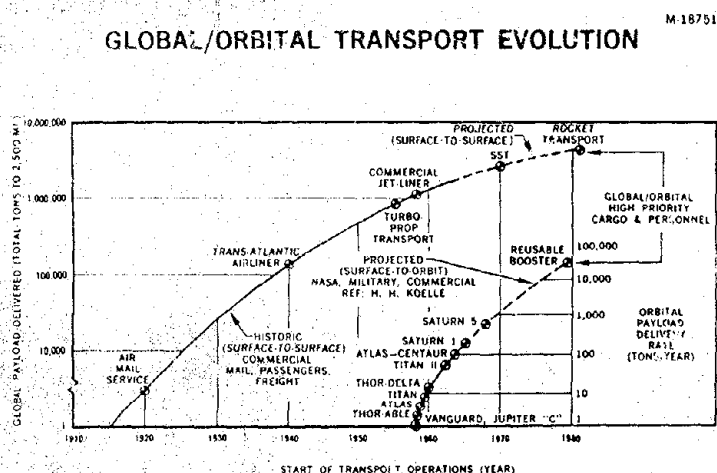


Figure 1

During the past five years, the age of space-flight infancy, launch vehicle development has been predicated on a pattern which was largely influenced by the earlier missile launching technology. Although this pattern of development provided a timely launch capability consistent with the exploratory nature of the space age, it has become apparent that marked improvement in launch vehicles (and launch operations) will be required to allow effective exploration of space in an economical manner.

The lower curve of Figure 1 implies that orbital missions exhibit a probable correlation to Earth surface rapid transportation missions. Since velocity requirements for global ranges are grossly comparable to those for orbital missions, it is possible that both markets may be

served by a single transport system. However, the traffic volume for global markets would exceed by several orders of magnitude that predicted for orbital traffic; thus, the effect on orbital operational costs could be profound. Only a highly ambitious launch program can justify the initial investment required for development of the Pegasus vehicle. Conversely, the existence in the national inventory of a highly versatile orbital/global rocket transport would undoubtedly stimulate innumerable uses for such a vehicle.

KEY DESIGN FEATURES

One of the most important predictions required as a basis for postulating any transport system is that of the optimum size of the vehicle to provide maximum mission versatility. Because the Pegasus study constraints imposed by funding and time limitations did not permit an extensive payload-capability synthesis and evaluation, an initial payload capability comparable to the Saturn V vehicle was adopted, although scalability to a smaller vehicle or offloading of propellants for smaller payload missions were maintained as major objectives during the Douglas-funded study. A reusable Saturn V vehicle could perform any Post-Apollo missions with significantly improved economy. During the Pegasus study, booster reusability, single-stage-to-orbit capability, and land recovery emerged as the predominant design and operational features for the conceptual configuration.

Since it appears logical to assume that the orbital carrier would be developed before its global counterpart, the subsequent material contained in this paper is subdivided into two major categories: (1) vehicle definition of the hypothetical Pegasus orbital booster (PEGASUS/OB) which employs ballistic (11 g's) atmospheric entry and (2) definition of the conjectural lifting-body global transport (PEGASUS/GT) derivative, which does not subject its passengers to more than 3 g's during ascent or entry.

THE PEGASUS SINGLE-STAGE-TO ORBIT VEHICLE

Figure 2 illustrates the reductions in size and weight which might be acquired by the use of advanced techniques. The two-stage expendable Saturn V vehicle on the left is compared with the single-stage-to-orbit Pegasus booster, a completely reusable system. The same 110-ft.-long Apollo payload is shown on top of each configuration to illustrate how advanced technology could reduce the overall length of the launch vehicle by 162 ft. with a corresponding reduction

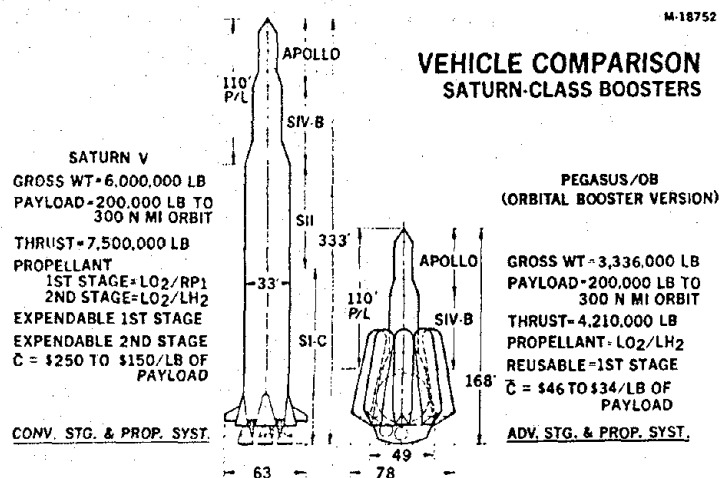


Figure 2

in gross weight of 2.66 million lb. The most significant aspect of this weight decrease is the reduction of the corresponding liftoff thrust and the associated reduction of acoustic effects on personnel and structure. It should be emphasized that each of these vehicles performs exactly the same missions; that is, each booster is sized to inject a 200,000-lb. payload into a 300-n. mi. orbit.

The most unconventional feature of the Pegasus vehicle is its reusability. Through this means, the cost per pound of payload to orbit could be reduced significantly. Figure 2 was not prepared as a critique on today's tendency to build larger ICBM's for space exploration but to demonstrate what potential gains might be realized through imaginative approaches to launch vehicle design. However, it must be remembered that these radical departures would necessitate expenditure of enormous sums, particularly for development of advanced propulsion systems. The payoff lies in long-range economies.

PEGASUS/OB, the orbital booster version of our reusable booster, is a one-stage-to-orbit disposable-tank vehicle utilizing liquid oxygen and liquid hydrogen as propellant and an annular plug-nozzle engine. Eight disposable tanks (used for liquid hydrogen only) are dropped during ascent to orbit and all tanks are recovered. After payload separation, the main vehicle is ejected from orbit by the same engine used for ascent. Propellant for orbital injection, rejection, and landing is contained within the main vehicle. Drag deceleration during atmospheric entry is accomplished in a base-first attitude. Aerodynamic heating of the plug nozzle is controlled by circulating liquid hydrogen through the plug to provide cooling. After terminal velocity is reached by the entry body, the main engines are reignited for retrothrust, and the vehicle sets

down on its four-legged landing gear. Hence, land recovery on Earth will exploit the technology which will be developed for the Apollo manned lunar landing.

Figure 3 presents a detailed description of the Pegasus orbital booster configuration. As indicated, the vehicle was designed to incorporate maximum use of a Pratt & Whitney modified RL-20 engine. That is to say, the oxidizer/fuel mixture ratio is compatible with this engine, as is the thrust level, the expansion ratio (of the hard nozzle), and the combustion chamber pressure. The Pegasus centerbody was sized with a 33-ft. upper diameter in order to accommodate the Saturn S-II or S-IVB stages and the Apollo payload. In this manner, the vehicle has been provided with sufficient flexibility to perform future Saturn V-type missions with greatly improved economy.

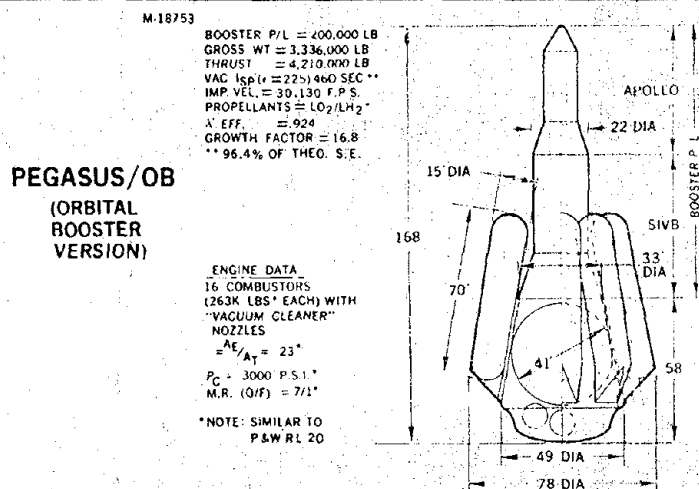


Figure 3

PROPULSION SYSTEM DEFINITION

With the recent emphasis on tripropellants and additives as methods of increasing I_{sp} of chemical propulsion systems, it appears that the potential gains available through extremely large expansion ratios have been completely overlooked. Figure 4 presents a plot of the vacuum specific impulse as a function of the nozzle expansion ratio. This expansion ratio is an effective value which can be acquired when the entire vehicle base can be used as the nozzle exit plane area. This condition can be satisfied by the use of conventional engine modules in an unconventional manner, that is, arranged peripherally around a centerbody. The centerbody would be contoured to act as a plug nozzle, providing altitude compensation to the vehicle. This approach would result in an increased expansion ratio, allowing the entire base to act as the nozzle exit area under vacuum conditions. At an expansion

LO₂/LH₂ ENGINE PERFORMANCE

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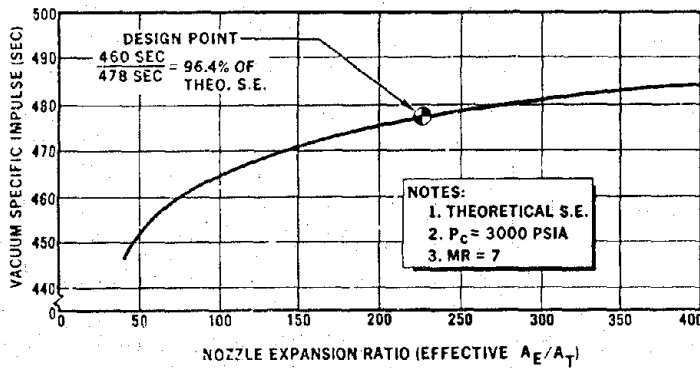


Figure 4

ratio of 225:1, a theoretical maximum I_{sp} of 478 sec. is possible. Sizing of the Pegasus vehicle was based on a 460-sec. value to allow for combustion and nozzle inefficiencies. It must be stated, however, that at these extremely high chamber pressures, frozen equilibrium conditions and reduced I_{sp} may govern the performance of the engine rather than the shifting equilibrium conditions shown. Extensive testing is required before the magnitude of this problem area can be resolved.

Figure 5 depicts two possible arrangements of thrust chambers around the centerbody plug. The left side of the figure indicates the excessive number of modules which would be required to completely surround the centerbody. Only a minimal gap could be tolerated between each bell nozzle exit lip to prevent shock interaction between the individual exhaust plumes, thereby avoiding the performance loss that would otherwise occur. The right side of the figure illustrates how these requirements can be met with considerably fewer modules, when each combustor is adapted with a vacuum cleaner type of

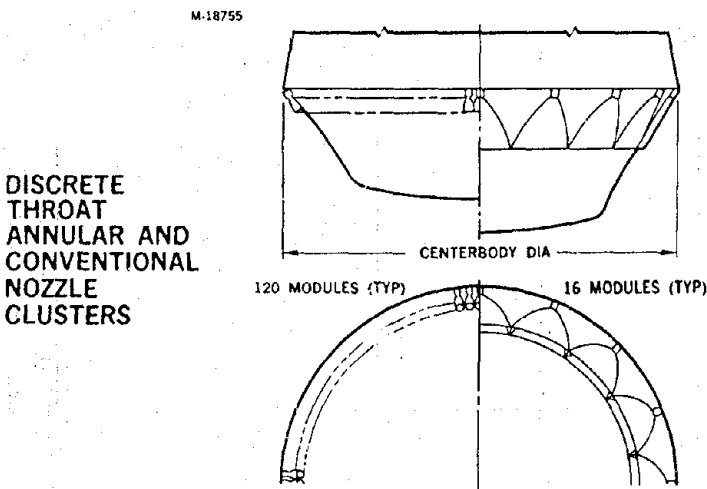


Figure 5

nozzle. This nozzle has a circular cross-section at the combustor throat and goes through a gradual transition to a curved oval shape at its exit plane. Aerojet-General has recently analyzed such a nozzle. The Rocketdyne toroidal combustion chamber also can be readily adapted to the Pegasus plug nozzle.

Figure 6 illustrates how the 50-ft. base diameter of the Pegasus configuration would require a vacuum cleaner nozzle approximately one and one-half times as wide as the diameter of the throat in order to provide an expansion ratio of 225:1. The base can then be completely surrounded with only 16 modules. By comparison, 120 conventional bell nozzles would be required to surround the same base. As depicted in the figure, the hard portion of the nozzle would be designed for a 23:1 expansion ratio.

GEOMETRICAL COMPARISON OF NOZZLE CONFIGURATIONS

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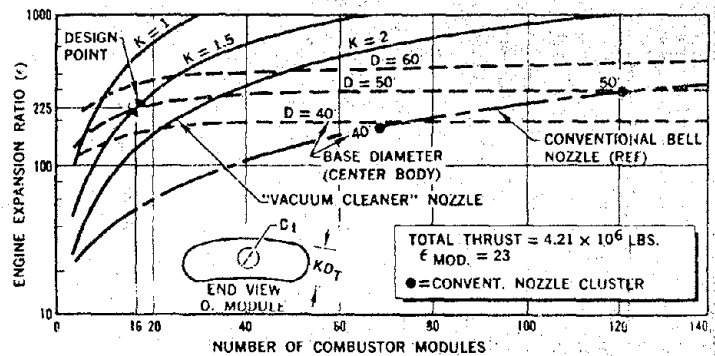


Figure 6

SINGLE-STAGE-TO-ORBIT SENSITIVITY

Figure 7 illustrates how, with today's specific impulse values, a single-stage-to-orbit vehicle would prove to be extremely sensitive to any reduction in the propellant mass-fraction since

GROWTH FACTOR SENSITIVITY TYPICAL ONE-STAGE-TO-ORBIT (LO₂/LH₂)

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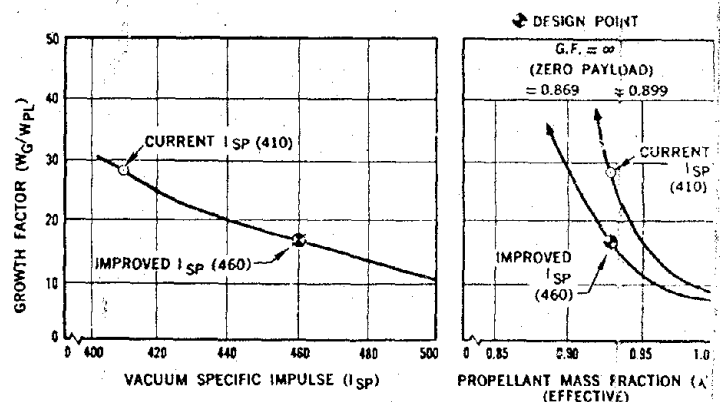


Figure 7

small decrease in the mass fraction could result in extremely large degradations of payload capability. If the vehicle mass fraction is inadvertently reduced to about 0.90, the growth factor becomes infinite. The vehicle then has no remaining payload capability. The specific impulse of 410 is a representative value when an engine with a conventional bell nozzle must operate during the complete ascent phase to orbit. In order to avoid flow separation at sea level, the expansion ratio of such a bell nozzle must be compromised at approximately 16:1, resulting in an extremely low-vacuum specific impulse value. With the altitude-compensating nozzle for increasing I_{sp} , the vehicle is no longer as sensitive to reductions in effective mass fraction (λ'). In addition, the latter specific impulse improvement can reduce the vehicle growth factor significantly.

The Pegasus vehicle is based on many radical departures from today's conventional booster operations. Figure 8 delineates some of the major advantages and disadvantages of one such departure. The principal advantage of the disposable tank feature is its effective mass fraction improvement and performance benefits which can be acquired during the ascent phase, when a portion of the inert weight is disposed of as soon as it is no longer needed. Perhaps equally as important as the ascent aspect is the recovery consideration. The entry maneuver can be performed far more effectively when a large portion of the air-frame weight (and bulk) is not recovered along with the main vehicle.

WHY DISPOSABLE TANKS?

ADVANTAGES

- IMPROVED PERFORMANCE DUE TO LOWER WEIGHT
- REDUCED FUEL HAZARD DURING ABORTED MISSIONS
- MORE RAPID PROPELLANT DUMP DURING ABORT
- REDUCED W/C_{DA} AND WEIGHT OF RECOVERY SYSTEM
- REDUCED SIZE AND WEIGHT DURING RETRIEVAL
- REDUCED OVERALL LENGTH OF LAUNCH VEHICLE
- SIMPLIFIED RECOVERY OF TANKS ALONE (W/O ENGINES)
- INCREASED MISSION FLEXIBILITY (OFF-LOADING)
- SIMPLIFIED ORBITAL REFUELING OPERATIONS

DISADVANTAGES

- OCEAN RETRIEVAL
- POTENTIAL LAND-MASS IMPACT
- SALT WATER CONTAMINATION
- INCREASED COMPLEXITY

Figure 8

LAUNCH VEHICLE RELIABILITY

Figure 9 presents an historical record of all U.S. space launches since November 1953. In order to give a valid sample, the more successful space launch vehicles, such as the Thor family (including Delta), are tabulated, as well as the less impressive Scout launches. As can readily

HISTORY OF U. S. SPACE LAUNCHINGS

(EXCL. ICBM'S - AS OF NOV. 1963)

LAUNCH VEHICLE	TOTAL VEHICLE LAUNCHES	BOOSTER 1 & 2 VEHICLE SUCCESSES (3 FAIL)	UPPER STG. VEHICLE SUCCESSES (5 FAIL)	BOOSTER RELIABILITY 8 SUCC./11 ATT. = 0.727	UPPER STAGE RELIABILITY 3 SUCC./8 ATT. = 0.375	VEHICLE RELIABILITY 3 SUCC./11 ATT. = 0.273
VANGUARD	11	8	3			
JUPITER C	6	6	3	1.000	0.500	0.500
JUNO II	10	8	4	0.800	0.500	0.400
THOR (INCL. AGENA, ABLE & DELTA)	119	111	94	0.933	0.847	0.790
ATLAS (INCL. AGENA & MERCURY)	42*	35	31	0.897	0.886	0.735*
SCOUT	15*	10	8	0.909	0.800	0.500*
TOTAL	204	178	143	0.908	0.803	0.701

*NOTE: INCL. 3 ATLAS AND 5 SCOUT FAIL (STAGE MALFUNC. NOT ANNOUNCED).

REF: STL SPACE LOG DATED DEC. 1963

Figure 9

be seen, the booster reliability for Thor has been a very impressive 0.933, for a sample of 119 launchings. The first stage of the Scout vehicle has an equally impressive reliability record. However, the solid-propellant Scout, which promised extremely high reliabilities and attractive cost-effectiveness at the outset of the program, has resulted in an overall launch vehicle reliability of only 0.50, when upper-stage malfunctions are considered. Perhaps this overall comparison (of booster versus launch vehicle reliability) provides the principal argument for single-stage-to-orbit vehicles, which would then avoid altitude start of engines (or stage separation) during ascent.

Accordingly, Figure 10 presents an estimated reliability apportionment of the Pegasus one-stage orbital booster and compares this reliability with that postulated for a hypothetical two-stage-to-orbit vehicle. A launch reliability of 0.96 was calculated for the single-stage-to-orbit booster. The overall mission reliability is reduced to approximately 0.935 when the malfunction probabilities of the recovery system are included. The Pegasus configuration dis-

ESTIMATED RELIABILITY PEGASUS/OB VS. PEGASUS/OB PLUS 2ND STAGE)

SYSTEM	CONCEPTUAL PEGASUS BOOSTER (SINGLE-STAGE-TO-ORBIT)	HYPOTHETICAL 2-STAGE ORBITAL VEHICLE
MAIN PROPULSION (INCL. T.V.C.)	0.984*	0.968*
REACTION CONTROL PROPULSION	0.998	0.982
ELECTRICAL	0.994	0.987
GUIDANCE	0.991**	0.976**
CONTROL	0.999	0.993
STRUCTURE	0.997	0.995
SEPARATION	0.996	0.995
LAUNCH VEHICLE	0.96*** (0.935 INCLUDING RECOVERY)	0.90

*REF: ROCKETDYNE RELIABILITY STUDY "BCI 62-48" INDICATES 30% OF ALL ENGINE MALFUNCTIONS OCCUR AT IGNITION 17% DURING ENGINE OPERATION (W/O SINGLE-ENGINE-OUT CAPABILITY) 3% AT ENGINE SHUTDOWN

** ASSUMES REDUNDANT GUIDANCE SYST. (PATH ADAPTIVE MODE) IN BOOSTER STAGE

*** ASSUMES NO MAIN ENGINE OPERATION DURING 45 MIN COAST PERIOD TO ORBITAL ALTITUDE (175 N. MI.)

Figure 10

plays a significant improvement in launch reliability over today's boosters since its multiple combustion chamber arrangement can be designed for the single-engine-out eventuality. Clearly, single-combustor boosters (Figure 9) could not incorporate such a feature.

As reflected in Figure 10, a recent reliability study indicates that checkout of all engine modules during the brief hold-down period on the launch pad will circumvent 80% of all engine malfunctions (which historically occur at engine ignition). Incorporation of the single-engine-out capability would prevent an additional 17% of all engine malfunctions, which occur during engine operation. This design philosophy would leave approximately 3% of all engine malfunctions unaccounted for, namely, those which occur at engine shutdown.

PAYLOAD FLEXIBILITY

The lefthand side of Figure 11 illustrates the hypothetical two-stage-to-orbit configuration referenced in Figure 10. The payload capability for this operational mode can be increased by 32,000 lb. in spite of the severe mass fraction degradation due to offloading of propellants. In addition to the improved payload capability of the two-stage configuration, that mode of operation may prove desirable during the flight test and development program in order to gradually build confidence in its eventual operation as a single-stage-to-orbit booster. The right side of Figure 11 illustrates the Pegasus configuration for a reduced-payload mission. As indicated, offloading of the propellant tanks for this mission would degrade the mass fraction from the nominal 0.924 value to approximately 0.892.

PEGASUS ORBITAL MISSIONS (PAYLOAD VERSATILITY)

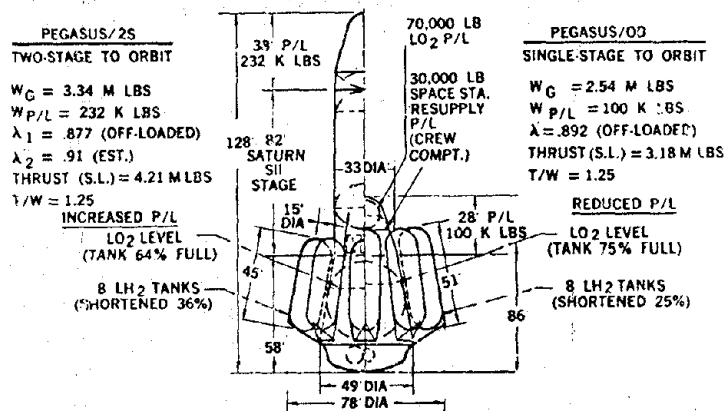


Figure 11

1990 WORLD'S FAIR IN ORBIT?

It has been speculated that a commercial market may develop in the distant future for taking high-income tourists to Earth-orbit and back. The rocket transport which carries its own oxidizer offers the most promising approach for satisfying such a future requirement.

Figure 12 presents a space station resupply payload requirement, based on the assumption of a 6-month duty cycle, and a 3-month resupply interval. Approximately 12 personnel and 18,500 lb. of consumables must be transported from the Earth's surface to orbit in order to recycle the 24-man crew of a space station, which may stay in orbit for five years. Although the early-1970 time scale estimated for this mission is not compatible with the availability of the Pegasus booster (1975 or later) it is used as representative of a typical orbital logistics requirement. For the design conditions stipulated, approximately 30,000 lb. of payload would be required during each launch--equivalent to the payload of one Saturn IB vehicle. Current launch philosophy with today's throwaway boosters would expend approximately \$346 million within a five-year period, merely to support space station operations.

ESTIMATED SPACE STATION RESUPPLY LOGISTICS PAYLOAD REQUIREMENTS

- $N_{PR} = N_{PT} (T_R / T_D)$
- ASSUMPTIONS
- SPACE STA. LAUNCHED BY SATURN 5 (EARLY 1970'S)
 - ORBITAL ALT. = 260 N. MI. (31.2° INCLINATION)
 - EASTERLY LAUNCH FROM AMR
 - RESUPPLY INTERVAL, $T_R = 3 \text{ MO.}$

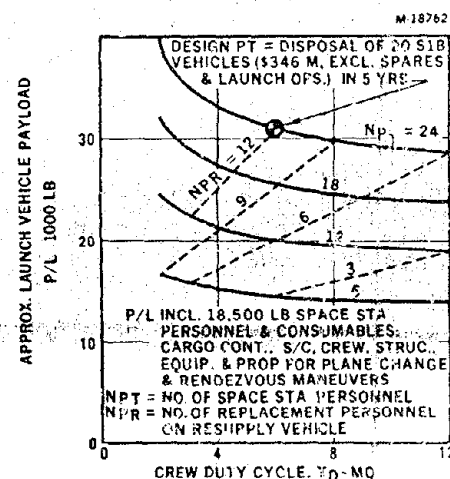


Figure 12

COST-EFFECTIVENESS OF ORBITAL BOOSTERS

Figure 13 compares costs of expendable and reusable single-stage-to-orbit vehicles, both of which consume liquid oxygen and liquid hydrogen propellants. It reflects the direct operating cost as a function of the nominal design payload of the vehicle. Assuming a launch rate of two per month (over a 10-year operational period), an expendable orbital booster with the 1-million-lb. payload of the post-Saturn class would cost ap-

D.O.C. COMPARISON OF SINGLE-STAGE- TO-ORBIT BOOSTERS

REUSABLE AND
EXPENDABLE
(CONSTANT LAUNCH RATE,
VARIABLE P/L)

- LAUNCH RATE = 2/MO
- TOTAL LAUNCHES = 240
- TEN YEARS OPERATIONS
- LO₂/LH₂ SYSTEMS

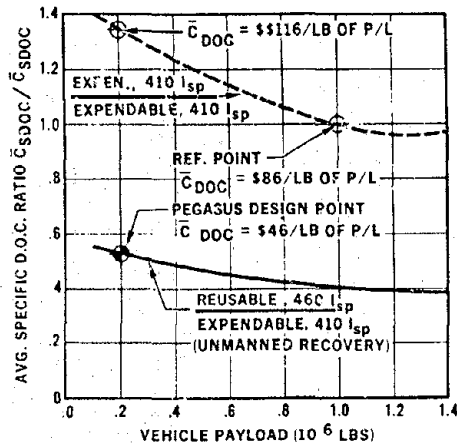


Figure 13

proximately \$86 per lb. of payload injected into orbit, based on current I_{sp} values. When the single-stage (expendable, low I_{sp}) vehicle size is reduced to a 200,000-lb-payload capability (similar to Pegasus), this direct operating cost would increase to \$116 per lb. of payload. For comparison purposes, the two-stage expendable Saturn V, using liquid oxygen and RP-1, would cost a minimum of \$150 per lb. of payload. When a single-stage vehicle incorporating advanced propulsion systems is initially designed for reusability, the cost can be reduced to approximately \$46 per lb. of payload. Moreover, it should be noted that the direct operating cost of the latter vehicle is far less sensitive to payload (and vehicle) size variations than its expendable, low-performance counterpart. A post-Saturn sized Pegasus (approximately 800,000-lb. payload) would cost \$37 per lb. of payload for the specified program of 24 launches per year.

Figure 14 is indicative of the further cost reductions which can be acquired through repeated reuse of a launch vehicle. The booster mission reliability is defined as the product of the launch reliability and the recovery system reliability. When the highly conservative launch reliability of 0.91 is adopted for Pegasus from the historical record of Figure 9, the resulting overall mission reliability of 0.89 would allow eight vehicle reuses, with a corresponding direct operating cost of \$46 per lb. of payload. When the estimated launch reliability (0.935) of Figure 10 is assumed, the related 14 reuses result in a direct operating cost of \$41 per lb. of payload, or an 11% cost reduction. Through repeated recovery, inspection, and improvements on subsystem malfunctions, mission reliability is certain to increase during a 10-year operational program. If that reliability should reach 0.975, allowing 39 vehicle reuses, the direct operating cost can be

PEGASUS/O.B. COST EFFECTIVENESS SENSITIVITY TO NUMBER OF REUSES

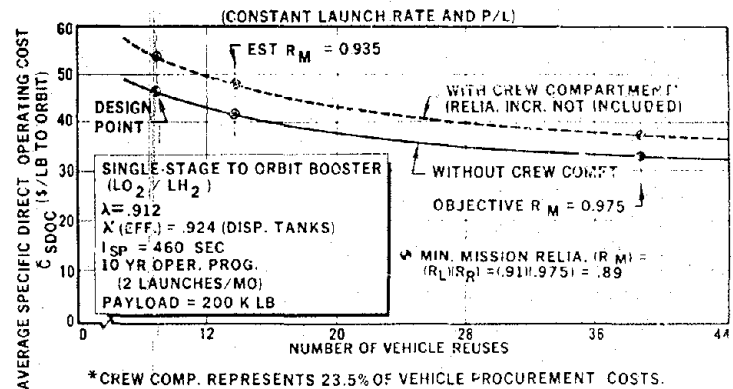


Figure 14

reduced to \$34 per lb. of payload. Under these conditions, the vehicle cost per mission is reduced to an insignificant level, and the total operating cost is comprised principally of launch operations and propellant costs. The increased vehicle (and operating) cost resulting from the incorporation of a two-man crew capsule for increased recovery probability, is reflected in the upper curve of Figure 14.

THE PEGASUS GLOBAL TRANSPORT VEHICLE

By comparing Figure 15 with Figure 3, it can be seen that the major distinction between the PEGASUS/GT (the global transport version) and the PEGASUS/OB (the reusable booster version) lies in the payload section configuration (which now contains the passenger compartment) and the addition of two fins, shown between the liquid hydrogen tanks. The fins are necessary to restrict the re-entry decelerations to a maximum of 2.5 g's. Without a lifting entry, ballistic decelerations would reach 10 or 11 g's; this might be acceptable for highly trained and conditioned

PEGASUS/GT (GLOBAL TRANSPORT VERSION)

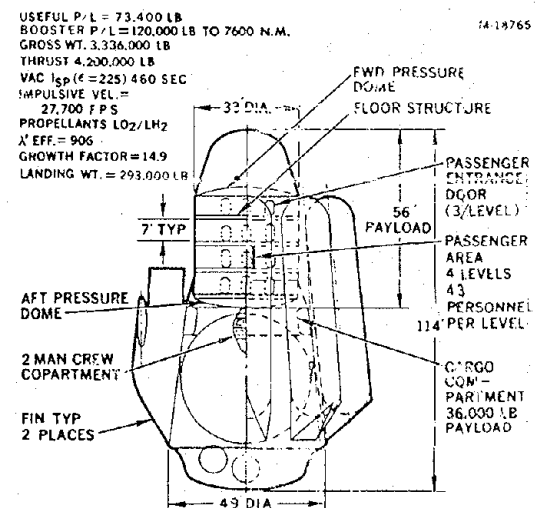


Figure 15

astronauts but would prove to be excessive for passengers, even those in the best physical condition. Therefore, a 3-g limit during boost and 2.5-g limit during entry were adopted as physiological criteria. During emergency entry conditions, the decelerations would be allowed to increase to a maximum of 3 g's.

The external liquid hydrogen tanks provide aerodynamic "shadowing" for the two fins during vertical (ballistic) ascent through turbulent atmosphere. The fins (with lateral control surfaces) are attached to the exterior of the centerbody, also providing lifting maneuverability for pinpointing the landing destination.

In addition to the physical differences of the two vehicles, the material (titanium) assumed for construction of the reusable booster would not withstand the increased temperature of the global transport entry mode when the underside of the vehicle is subjected to a higher heat flux; therefore, stainless steel would replace portions of the titanium structure of the reusable booster.

The parent Pegasus reusable booster weighs 3.34 million lb. at liftoff. The same gross weight was adopted for the hypothetical global transport; hence, the same advanced engines, which would have to be developed for the orbital mission, would be directly applicable to the global transport version of the vehicle. Since PEGASUS/OB requires throttleability (or thrust modulation of engines) for its primary mode of operation, the PEGASUS/GT flight profile was based on the same propulsion system characteristics. A Saturn V class of reusable orbital booster (with a liftoff thrust of 4.2 million lb.) will provide a global transport derivative with the capability of transporting 73,400 lb. of useful payload to a maximum range of 7,600 n.mi. (assuming that launch sites are available on both the east and west coast of the continental United States).

VEHICLE DESCRIPTION

The floor plan of Figure 16 illustrates how the 43 individual passenger couches are arranged on each of the four decks. During the few minutes of weightless flight, the personnel are constrained by belts attached to the couches. On a rocket-propelled ballistic Pegasus mission the acceleration vector during ascent and the deceleration vector during entry are oriented in the same direction with respect to the personnel; that is, the acceleration loads would be distributed by the couch to the same portion of each passenger's back during either phase of the flight

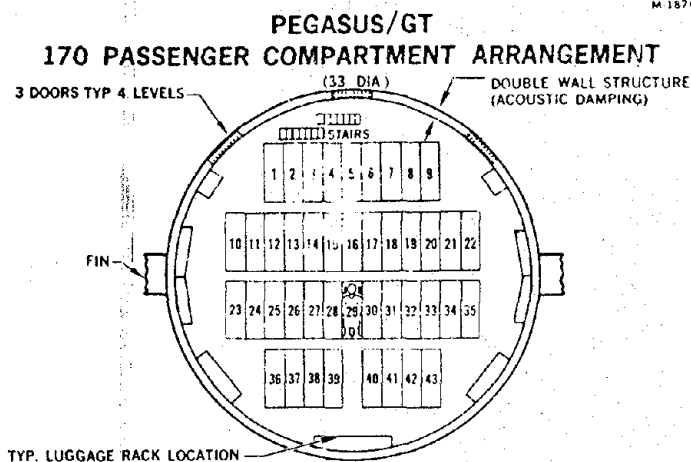


Figure 16

regime. By comparison, where the pilot and crew are facing forward for visibility prior to landing during a glide-type re-entry, they are subjected to severe discomfort and handicap from the deceleration vector orientation (eyeballs protruded).

The floor plan illustrates the proposed rack locations and access stairwells which interconnect the four decks for emergency egress. Three entry doors are located at each of the four deck levels for rapid loading and unloading of passengers. To diminish the noise effect on personnel, the 33-ft.-dia. payload compartment would be constructed of a double-wall, acoustic-damping material. Prior to loading of passengers, the vehicle would be completely checked out, and the propellant tanks would be filled. Constant topping would ensure that the propellant in the tanks was at the proper level prior to liftoff.

The passengers would enter the vehicle through a tower that incorporates a number of elevator platforms which lead to the loading ramps. Three ramps, located between the external hydrogen tanks, service the entry doors at each of the four deck levels of the vehicle payload. During atmospheric entry, these door openings are located on the upper side of the vehicle, thus keeping the highly heated underside free of structural openings and discontinuities.

Figure 17 compares the principal design parameters of the two vehicles. One major distinction between the concepts can be noted by the comparative values of the effective mass fraction (λ'). The designation "effective" indicates that the benefits attendant with disposal of hydrogen tanks during flight have been included in the performance calculations. The reusable booster effec-

PEGASUS/ORBITAL BOOSTER GLOBAL TRANSPORT COMPARISON

M-18767

	PEGASUS/OB	PEGASUS/GT
	HYP. NASA BOOSTER (P/L = 200 K LB TO 300 N.MI ORBIT)	ROCKET TRANSPORT (P/L = 120 K LB TO 7600 N.MI RANGE)
MAX. THRUST (LB)	4.2 M	4.2 M
THRUST-TO-WT. RATIO (T/W)	1.25*	1.25*
MAX. LIFT-OFF WT. (LB)	3.3 M	3.3 M
USABLE PROPELLANT (LB)	2.9 M	2.9 M
NOZZLE EXPANSION RATIO (ε)	225	225
VAC. SPECIFIC IMPULSE (SEC)	460***	460***
EFF. MASS FRACT. (λ)	.924	.910
IMPULSIVE VELOCITY (F.P.S.)	30.1 K	30.9 K
LANDING WEIGHT (LB)	150 K	305 K
OVERALL LENGTH (FT)	168 (58 REC.)	114
PAYLOAD DIA (FT)	22	33

*CONSTANT FOR ALL ORBITAL PAYLOADS AND ALL MISSION RANGES
**BOOSTER P/L = 120K LB. USEFUL P/L = 73K LB
***LO₂/LH₂ PROPELLANT SYSTEM
P_c = 3000 PSI. M.R. (O/F) = 7/1

Figure 17

tive mass fraction has been calculated, after rigorous analysis, to be 0.924. The mass fraction of PEGASUS/GT is reduced to allow for, (1) the added structural weight of its fins, (2) the added weight of the stainless steel which replaces the titanium structure on the underside of the vehicle, (3) the added liquid hydrogen required for cooling the blunt nose during entry, (4) the increased retropropellant required, and (5) the added strength required of the four landing legs and attach fittings. The PEGASUS/GT vehicle lands with payload attached, while the PEGASUS/OB reusable booster separates its payload in orbit. On the latter vehicle, centerbody (without payload) is landed.

The transport version, although not acquiring orbit, requires some 800 fps in excess of orbital velocity for its ballistic mission because of the increased gravity losses during vertical ascent. However, since the magnitude of the two mission velocity requirements are grossly comparable, it is possible to directly adapt a booster, initially sized for a single-stage-to-orbit mission, to a ballistic global transport mission.

ACOUSTIC EFFECTS ON PERSONNEL AND STRUCTURE

One of the major problem areas presented by a Pegasus type of operation is associated with the noise generated by the engine. Figure 18 compares two estimated acoustic bands with one measured during recent Saturn I launches. As shown, a measured value of 167 db was recorded near the nozzle exit plane of the Saturn I vehicle, with its 1.5 million lb. of thrust, and liquid oxygen/RP-1 propellants. The middle band of the figure defines the predicted range for the Pegasus vehicle with its 4.2 million lb. of thrust, plotted as a function of the distance forward of the nozzle

PREDICTED OVERALL SOUND PRESSURE LEVEL (OASPL) (NEGLECTING PRESSURE DOUBLING)

M-18768

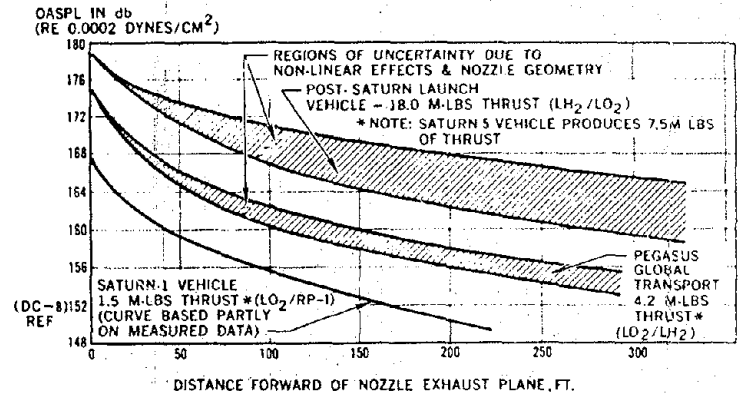


Figure 18

exit plane. In the vicinity of the engine, it can be seen that approximately 175 db. are predicted for this thrust level and propellant combination. The upper band indicates that approximately 179 db are predicted for the post-Saturn class of launch vehicles, assuming liquid oxygen/liquid hydrogen propellants and a thrust level of 18 million lb. For the sake of comparison, the DC-8 produces a noise level of 152 db. Conspicuous by its absence from this figure is the noise level prediction for the oxygen/RP-1 Saturn V vehicle, with its 7.5 million lb. of thrust; it is expected that this acoustic excitation will lie somewhat above the calculated values for Pegasus but considerably below the band defined for the post-Saturn class of vehicle.

Figure 19 enumerates some techniques which may prove effective in reducing the anticipated noise levels to within tolerable limits. A rigorous investigation has not yet been conducted to assess the intensity of acoustic energy imposed on the passenger compartment of the vehicle. Nevertheless, it appears that some supplementary techniques must be employed to further attenuate the acoustic excitation on the

PEGASUS NOISE REDUCTION TECHNIQUES

M-18769

- REDUCTION OF NOISE AT SOURCE
 - AIR AUGMENTATION OF JET STREAM
 - OPTIMIZE LAUNCH PAD DESIGN (REFLECTED ACOUSTIC ENERGY)
 - DUCTING (DEFLECTION) OF JET STREAM
 - IMPINGEMENT OF JET ON WATER SURFACE
- REDUCTION OF NOISE IN PASSENGER COMPARTMENT
 - MECHANICAL ISOLATION OF FLOOR SUPPORTS AND SEATS
 - LOW-MOLECULAR WEIGHT GASES (OR VACUUM) BETWEEN MULTIPLE WALLS
 - INCREASED MASS OF INNER WALL
 - INCORPORATE PANELS OF ABSORPTIVE MATERIALS
 - REDUCTION OF INTERNAL (CABIN) PRESSURE
 - REDUCTION OF MOLECULAR WEIGHT OF BREATHING GASES (HELIUM)
- REDUCTION OF NOISE ON PERSONNEL
 - ISOLATION OF EARS (EAR PLUGS, EAR MUFFS)

Figure 19

passengers. One or more of the techniques tabulated may effectively accomplish this purpose. The noise problem must be resolved before consideration can be given to the transportation of personnel by rocket-powered vehicles.

WEIGHT CONSIDERATIONS

Figure 20 presents a breakdown of the Pegasus vehicle inert weight, which amounts to approximately 326,000 lb. Approximately 20% of this inert weight can be charged directly to the recovery system, which includes the propellant for cooling the plug nose during atmospheric entry and also the propellant for providing retrothrust and hover capability. It is interesting to note that an equivalent inert weight is jettisoned when the external liquid hydrogen tanks are discarded. Analytical results indicate that, in terms of increased payload, the technique of tank disposal in flight more than compensates for recovery system weight. Also of interest is the fact that the control fins weigh more than half of the centerbody weight. Included in the inerts are the flight propellant reserves which are necessary to provide the vehicle with a velocity capability 1% beyond the normal mission requirements.

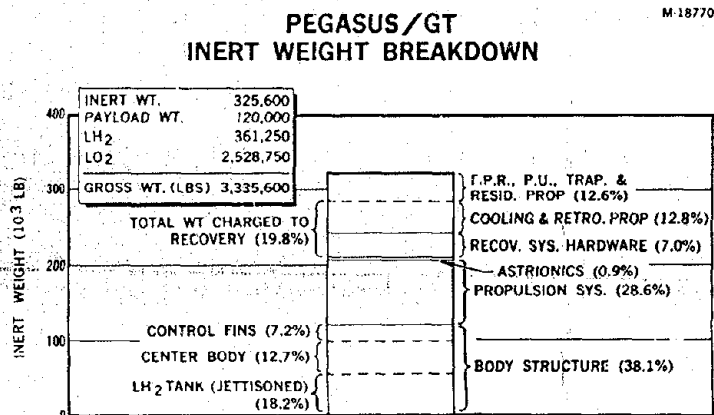


Figure 20

Figure 21 tabulates the PEGASUS/GT payload weight. The term "booster payload" (120,000 lb.) is a carry-over from the orbital booster analysis and is equivalent to 73,400 lb. of usable payload to a 7,600 n.mi. range. Approximately 220 lb. per man were allocated for each of the 170 passengers, including 40 lb. of personal luggage. An appreciable allotment of 3,200 lb. was included for the weight of gas to pressurize the huge volume of the passenger compartment. A 6,000-lb. allocation was deducted directly from the useful payload to compensate for the weight

PEGASUS/GT PAYLOAD PRELIMINARY WEIGHT ESTIMATE

M-18771

*PASSENGERS (170 PERSONNEL AT 180 LB/PERSON)	30,600 (LB.)
*PASSENGER BAGGAGE (40 LB/PERSON)	6,800
PASSENGER PROVISIONS (20 LB/PERSON - SEATS, RESTRAINTS)	3,400
*CARGO PAYLOAD (18 TONS)	36,000
CABIN PRESSURIZATION SYSTEM AND GAS (12 PSIA)	3,200
CABIN STRUCTURE (CYLINDRICAL SIDEWALL, PRESS. BULKHEAD, AND FLOORING)	22,000
ACOUSTIC DAMPING PROVISIONS	5,000
NOSE FAIRING	7,000
CREW SYSTEM (2 CREWMEN, CAPSULE, ESCAPE SYSTEM, **ENVIRONMENTAL SYSTEM, ETC.)	6,000
BOOSTER PAYLOAD TO 7600 N.MI. RANGE	120,000 (LB.)
*USEFUL PAYLOAD TO 7600 N.MI. RANGE = 73,000 LB	
** FLIGHT TEST ONLY	

Figure 21

of the two-man crew capsule, escape system, and environmental control system. Many of the preliminary weight estimates shown for structural components can be significantly reduced through more rigorous design refinements.

GLOBAL MISSION PROFILE

With the assumption of two launch-pad locations (one each at the Atlantic and Pacific Missile Ranges), a range halfway around the world (10,800 mi.) is not necessary to reach the key destinations which were assumed. Clearly, such a range also can be acquired by trading payload for added propellants. For the maximum-range missions (such as PMR to Singapore), the vehicle would be fully loaded with propellants. With the payload weight kept constant, propellants would be offloaded prior to liftoff for the shorter-range missions. For example, a mission from AMR to London would necessitate propellant tanks only 56% full. The engines are throttled at liftoff to maintain a constant thrust-to-weight ratio of 1.25, regardless of the destination. A vehicle which is designed with disposable tanks is readily adapted for an offloaded mission merely by the elimination of some of the external propellant tanks. The internal oxygen tank would not be filled to maximum capacity.

Figure 22 tabulates the transit times of the Pegasus vehicle to 14 principal cities and compares these times with those of modern jet transports and the proposed supersonic transport (assumed to travel at Mach 2.5). It should be noted that neither the supersonic transport nor the jet transport will possess the extensive range capability of a Pegasus vehicle. Assuming that the supersonic transport could travel the 7,500 n.mi. from PMR to Singapore without stopping to refuel, its flight duration would be

M-18772 **BALLISTIC TRANSPORT CAPABILITY ESTIMATE TO PRINCIPAL CITIES OF THE WORLD**

DESTINATION	LAUNCH SITE	GLOBAL DISTANCE (N. MI.)	VEHICLE PROPELLANT LOADING (% FULL)	EST. TRANSIT TIME		
				GLOBAL TRANSPORT (ROCKET) (MIN.)	SUPERSONIC TRANSPORT (MACH 2.5) (HR.)	JET TRANSPORT (V=550 MPH) (HR.)
1. DAKAR, SENEGAL	AMR	3550	54	25	2.1	6.5
2. LONDON, ENGLAND	AMR	3750	56	26	2.2	6.8
3. RIO DE JANEIRO, BR	AMR	3810	57	26	2.2	6.9
4. BUENOS AIRES, ARG	AMR	3970	58	26	2.3	7.2
5. MOSCOW, USSR	AMR	4850	64	30	2.9	8.8
6. CAIRO, EGYPT	AMR	5550	69	33	3.3	10.1
7. CAPE TOWN, SO. AF	AMR	6510	76	36	3.8	11.8
8. BOMBAY, INDIA	AMR	7100	98	39	4.4	13.5
9. FAIRBANKS, ALA	PMR	2030	50	17	1.2	3.7
10. HONOLULU, HAWAII	PMR	2140	52	18	1.3	3.9
11. TOKYO, JAPAN	PMR	4640	77	30	2.7	8.4
12. NOVOSIBIRSK, USSR	PMR	5270	87	32	3.1	9.5
13. MELBOURNE, AUS	PMR	6810	96	37	4.0	12.3
14. SINGAPORE, MAL	PMR	7510	100	39	4.4	13.5

PEGASUS P/L = 120K LB. (USEFUL P/L = 73K LB.) MAX. RANGE = 7600 N. MI.

Figure 22

seven times as long as that required by Pegasus. The jet transport would require a travel time approximately 20 times greater.

Figure 23 gives a preliminary route structure and estimated transit times between 16 key cities, most of which are major seaports. A minimum distance of 2,000 n.mi. between cities was assumed. These coastline cities were selected because a majority of them are considered by commercial cargo carriers to be locations for global network coverage. Rocket transportation between each of these locations would involve water overfly during vertical liftoff from the launch site. In the event of unavailable land mass for a spaceport at a densely populated city, off-shore landing and launch facilities might be considered.

M-18773 **ASSUMED GLOBAL TRANSPORT NETWORK EST. TRANSIT TIME (MIN.)**

	BUENOS AIRES, ARG.	BOMBAY, INDIA	CAIRO, EGYPT	CAPE KENNEDY, U.S.A.	CAPE TOWN, SO. AFRICA	DAKAR, SENEGAL	HONOLULU, HAWAII	LONDON, ENGLAND	MELBOURNE, AUSTRALIA	MURMANSK, U.S.S.R.	NOME, ALASKA	RIO DE JANEIRO, BRAZIL	SINGAPORE, MALAYA	TOKYO, JAPAN	VANDENBURG, U.S.A.	VLADIVOSTOK, U.S.S.R.
BUENOS AIRES, ARG.				26												
BOMBAY, INDIA				39												
CAIRO, EGYPT				33												
CAPE KENNEDY, U.S.A.	26	39	33		36	25		26		32		26		32		29
CAPE TOWN, SO. AF.				35					36		30					
DAKAR, SENEGAL				25												
HONOLULU, HAWAII																
LONDON, ENGLAND	39	30		26										38		18
MELBOURNE, AUS.					36											33
MURMANSK, U.S.S.R.				32	39	30						39				34
NOME, ALASKA													35			21
RIO DE JANEIRO, BR.									39							39
SINGAPORE, MAL.																30
TOKYO, JAPAN		29														33
VANDENBURG, U.S.A.														39		
VLADIVOSTOK, U.S.S.R.							18		37	21			39	30		

Figure 23

Figure 24 illustrates a typical mission profile which would result from an imposed 3-g limitation during boost and entry. Booster burnout would occur approximately 6 min. after liftoff

M-18774 **PEGASUS/GT MISSION PROFILE—WESTERLY LAUNCH (MAX. RANGE—PMR TO SINGAPORE) 3-g LIMITED ASCENT & DESCENT TRAJECTORY**

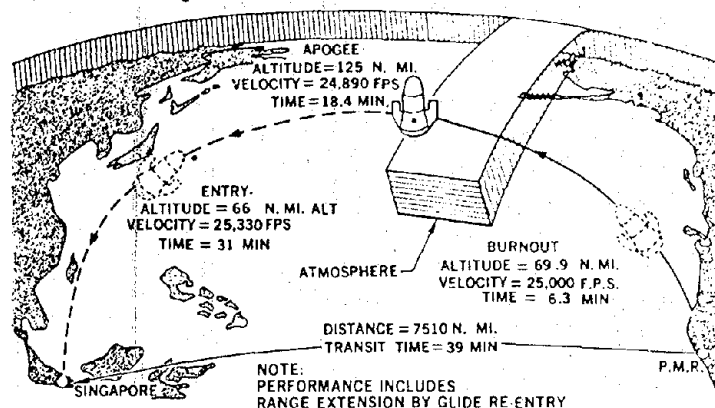


Figure 24

at an altitude of 70 n.mi. and a velocity of 25,000 fps. The vehicle would coast for an additional 12 min. until apogee was acquired. At this point, the velocity has decreased to 24,900 fps. The apogee altitude of 125 n.mi. is well above the edge of the sensible atmosphere. Hence, most of the mission would be accomplished where drag is nonexistent. Some 13 min. after the apogee condition, the vehicle will start the entry portion of the flight profile when it again approaches the edge of the atmosphere at an altitude of approximately 400,000 ft. The reaction control system will orient the vehicle into the required 49° nose-up attitude prior to entry.

At liftoff, due to the engine thrust-to-weight ratio, the vehicle is accelerated at 1.25 g's. As propellant is depleted, and the velocity would increase, the engines are throttled to restrict the vehicle to a maximum acceleration of 3 g's. This condition is maintained until main engine cutoff. During re-entry, the vehicle again is restricted to a 3-g maximum emergency condition (2.5-g nominal). The bank angle is modulated at constant altitude to satisfy the 3-g condition. A 52° angle-of-attack will produce a vehicle lift-to-drag ratio of approximately 0.42. After drag has cancelled the horizontal velocity, and the vehicle has reached a stalling condition, the attitude control system will orient the vehicle through an angle of 102° until the base is pointed directly downward. A few segments of the propulsion system are then restarted to nullify the vertical component of velocity. The vehicle has the capability of hovering and translating horizontally prior to settling down on the four extensible legs.

Mission Abort

During a normal ascent, Pegasus would rise almost vertically for about 70 sec. In the event of

an engine malfunction or an emergency abort condition (see Figure 25), the eight external tanks containing the hazardous liquid hydrogen can immediately be separated and jettisoned at sea. The major portion of the liquid oxygen would be pumped overboard from the internal tank through the turbine discharge port located in the center of the engine plug nozzle. In an aborted mission ample propellant would be retained on board to assure that adequate retro-thrust can allow the entire vehicle to soft land at sea. In this sense, a rocket-powered VTOL, which can use an entire ocean (or a small patch of unprepared terrain) as its emergency landing site, might be inherently safer than a jet aircraft which must reach a particular airport for an emergency landing.

PEGASUS/GT EMERGENCY PROPELLANT DUMPING (ABORTED LAUNCH)

M-18775

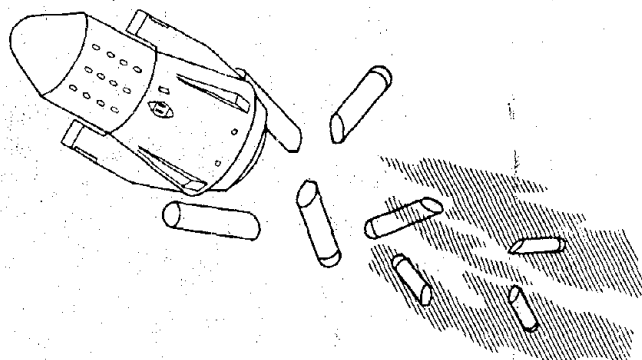


Figure 25

After propellants are dumped, four expandable-structure spheres (Figure 26) would automatically be deployed from the extended landing legs to assure hydrostatic stability of the entire vehicle after it alights on the ocean. Since only a few segments of the engine are required to brake the empty vehicle, those segments between landing legs would be selected. During this emergency recovery mode, the vehicle would be towed back to port. Numerous surface vessels and tow lines will stabilize the vehicle against adverse wind effects during the retrieval process.

Figure 27 presents a preliminary analysis of the aborted mission investigation. Three failure times were arbitrarily selected along the normal ascent trajectory. It was assumed that a malfunction might occur at 90, 130, and 195 sec. after liftoff. The three bands, associated with these failure times, reflect the uncertainty of downrange impact locations due to atmospheric drag effects. The figure shows that not much more than 10 or 20 sec. (after detection of a malfunction) might be available to jettison ap-

PEGASUS/GT LAUNCH ABORT (ALTERNATE RECOVERY MODE)

M-18776

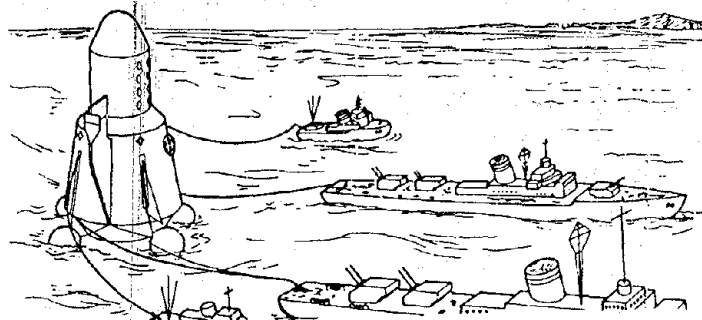


Figure 26

FLIGHT PATHS AND REACTION TIMES FOR ABORT MANEUVERS (BALLISTIC RETURN)

M-18777

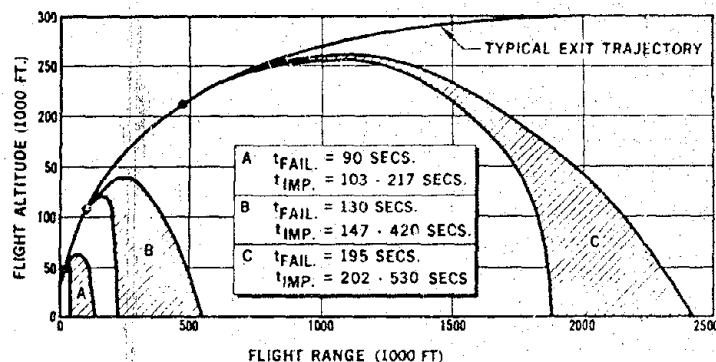


Figure 27

proximately 3 million lb. of onboard propellant. Clearly, the rapid ejection of the full LH₂ tanks is highly desirable in such an event. Studies are currently in progress to determine the most expeditious method of pumping the liquid oxygen overboard from the internal spherical tank.

Emergency Escape During Flight Test

Figure 28 illustrates the pressurized two-man crew compartment which would be installed within the booster centerbody, above the spherical liquid oxygen tank. The crew would enter this compartment by way of the external door and access ramp, and then through the airlock.

The airlock into the crew compartment provides mission flexibility, allowing the crew to participate in orbital rendezvous operations outside the PEGASUS/OB parent vehicle. The 6,000 lb. of this pressurized compartment has been incorporated into the weight estimate for the PEGASUS/GT vehicle (Figure 21). Redirection capability could be incorporated into the on-board computer

PEGASUS 2 MAN CREW COMPARTMENT

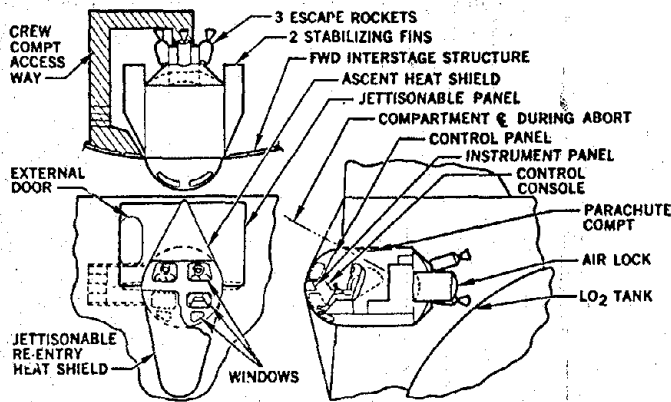


Figure 28

which controls vehicle guidance to predetermined destinations. The crew would be provided with manual override of the computer.

Three solid-propellant motors provide emergency escape capability for the crew during an aborted flight; however, it should be noted that this capability will only be used during the flight test and development phase of the program. During normal operations, the entire vehicle will have complete abort (water-recovery) capability.

Two heat shields are provided above and below the external portion of the crew compartment to protect the windows from excessive heating during ascent or entry. The lower heat shield is jettisoned after entry just before the terminal retro phase. The upper heat shield, which is necessary during ascent, is only jettisoned prior to an emergency abort of the crew capsule. For such an event, two stabilizing fins are incorporated on the crew compartment sides to prevent the capsule from tumbling during operation of the solid-motor escape rockets. The weight of ablatant, necessary for thermal protection of the fins and the compartment underside, has not yet been assessed.

Figure 29 depicts a typical low-altitude emergency escape condition, and illustrates the front panel being jettisoned along with the ascent heat shield. Prior to ejection, the centerline of the entire capsule is automatically rotated upward approximately 30°. The escape rockets then will thrust in an upward direction (as well as outboard), away from the jeopardized vehicle.

Figure 30 shows how the crew-capsule recovery parachutes are deployed from their stowage compartment at the appropriate altitude. In addition,

PEGASUS CREW CAPSULE EMERGENCY ESCAPE (FLIGHT TEST ONLY)

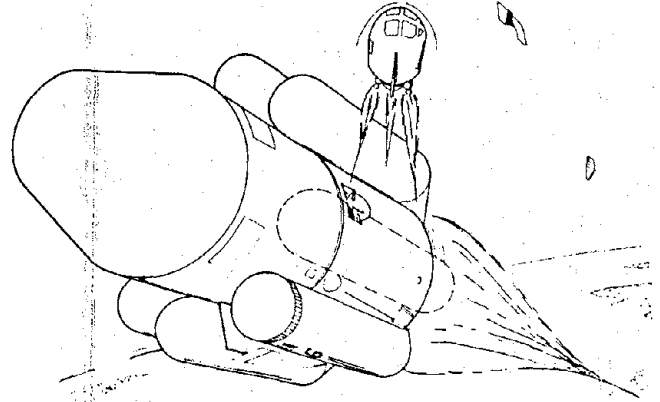


Figure 29

PEGASUS EMERGENCY CREW ABORT (FLIGHT TEST ONLY)

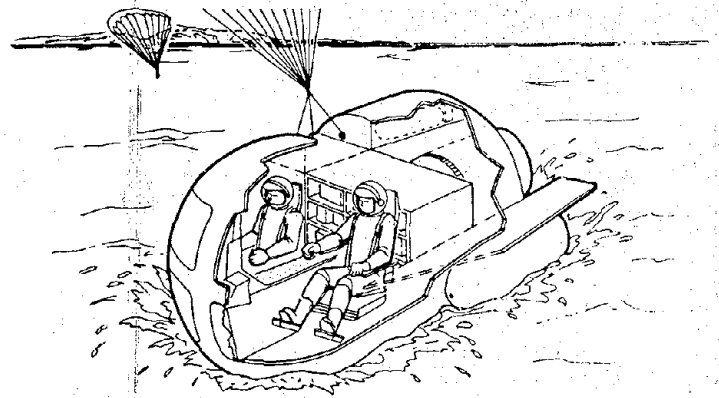


Figure 30

expandable pneumatic bags are deployed from beneath the stabilizing fins to provide roll stability after touchdown at sea and to absorb impact energy during a land recovery.

Parametric Mission Criteria

Figure 31 represents an historical plot of the ascent trajectory for a representative mission from AMR to London. The parameters of velocity, altitude, flight-path angle, and acceleration are plotted as a function of time and liftoff. The initial thrust-to-weight ratio of 1.25 will build up to approximately 3 g's after 150 sec. of flight. Acceleration is maintained at 3 g's for 3.5 min. until burnout by progressive throttling of the main engines. The plot defines the point at which the first four external hydrogen tanks are jettisoned, some 135 sec. after liftoff. The propellant is depleted from these four tanks concurrently. In order not to adversely affect the vehicle's stability, the tanks are separated simultaneously. Approximately 240 sec. after liftoff, the next pair of tanks are jettisoned; the last pair are ejected just after main engine cut-

PEGASUS/G.T.-ASCENT TRAJECTORY

M-18781

CAPE KENNEDY TO LONDON (EASTWARD LAUNCH-RANGE = 2750 N. MI.)

NOTES: 1. $W_G = 3.34 \times 10^6$ LB 2. $T/W_{LO} = 1.25$ 3. $t_{sp} = 377-460$ SEC

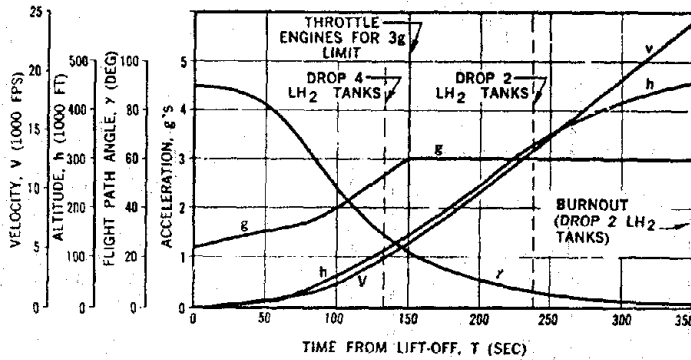


Figure 31

off, 350 sec. after liftoff. These tanks are lowered by parachute and recovered from the ocean by landing ship dock (LSD), as described in Reference 1.

During ascent, the control surfaces of the two fins (located between the hydrogen tanks) are stowed in the locked position. As the vehicle is boosted through the atmosphere, the trailing edge of these control surfaces will be subjected to negligible heat flux. One potential problem area, which has not been thoroughly investigated, is the degree of shock wave reflection between the external LH₂ tanks and the fin surfaces. The intensity of heating and buffeting which may result from these interactions should be resolved initially through wind tunnel testing.

Since a ballistic transport vehicle of the type described must have launch capability in an east or a west direction, a nonrotating Earth was assumed for the preliminary estimate of payload capability. As a matter of academic interest, Figure 32 illustrates the increase in payload which can be acquired during an easterly launch.

PEGASUS/GT ACTUAL PAYLOAD VS RANGE FOR 3-G LIMITED MISSION

(EASTWARDLY LAUNCH
FROM AMR)

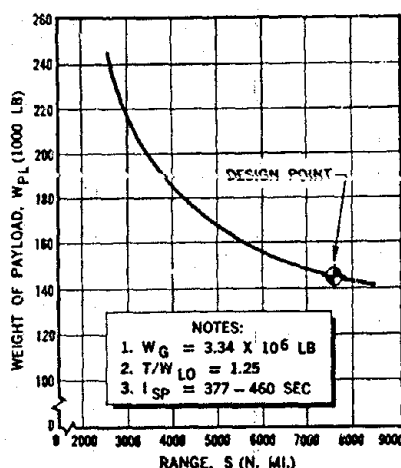


Figure 32

For example, the nominal 120,000-lb. booster payload (to a 7,600-n.mi. range) can be increased to 145,000 lb. during an easterly launch; added velocity is imparted to the vehicle by the Earth's rotation. The propulsion system specific impulse varies from 377 sec. at sea level to 460 sec. under vacuum conditions. These values are based on an assumed chamber pressure of 3,000 psi, a mixture ratio (oxygen to hydrogen) of 7:1, and an expansion ratio of 225:1 (provided by the altitude-compensating nozzle).

Figure 33 plots the atmospheric entry phase of the mission trajectory. When lift modulation is accomplished by variation of the bank angle during entry, deceleration limits can be restricted to a nominal peak of 2.5 g's (3 g's to allow for emergencies). After the initial peak deceleration, the physiological loads vary between 1.3 and 2.2 g's for the remainder of the entry maneuver. It should be noted, however, that the passengers would be subjected to more than 2 g's for only two brief periods of approximately 1.5 min. each. The personnel would then have available a 2-min. interval (for re-

PEGASUS/GT PRELIMINARY ENTRY TRAJECTORY BANK ANGLE-MODULATED ENTRY

M-18782

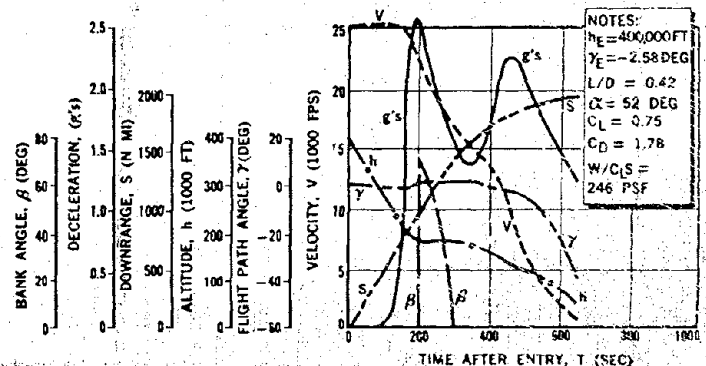


Figure 33

covery from 2 g's) prior to active debarkation. The peak deceleration load occurs 190 sec. after entry initiation. Shortly thereafter, the vehicle is banked through an angle of 70° (around the velocity vector), which is progressively decreased for 100 sec.

During this interval, the vehicle maintains a constant altitude of 180,000 ft. and a constant flight-path angle of 2.7° down from the local horizontal. Three minutes after entry into the edge of the atmosphere (400,000 ft.), the flight-path angle is decreased to 0°. The vehicle is maintained at a constant angle of attack of 52° from the velocity vector throughout the entire entry maneuver. At the end of the entry phase

(after 10.5 min.), the velocity has decayed to 1,000 fps at an altitude of 50,000 ft. (Mach 1.03). Shortly after flight through the transonic regime, aerodynamic drag will further reduce the horizontal component of the terminal velocity to a negligible value, and a quasi-stalled condition will result. The flaps are then brought to neutral position, allowing the vehicle to pitch over into a vertical engine-down attitude and start its free fall. Segments of the main engine are ignited at 2,500 ft. to cancel vertical velocity. At pitchover, the initial flight-path angle has decreased to 40° below the horizontal. The vehicle centerline is oriented 49° above the horizontal at the start of entry, and 12° above the horizontal at the end of atmospheric entry. It should be noted that the high wing loading of 246 psf would result in prohibitive touchdown speeds and thereby preclude a horizontal landing in the conventional manner.

Figure 34 delineates the ranges of deceleration to which many persons in average physiological condition are often subjected. The emergency braking system on a hotel elevator will subject passengers to 3 g's while in a standing position--an extremely uncomfortable position for withstanding decelerating loads. The arresting gear on an aircraft carrier deck subjects the plane's pilot to 4 g's for a very brief period. During the shipboard landing operation, the orientation of the deceleration vector would render the pilot ineffective to cope with an emergency situation which might arise. The figure shows that the snatch loads associated with parachute deployment can subject personnel to a short-term exposure as high as 15 g's.

Although the 3 g's adopted as a passenger limit during ascent and entry would prevent elderly persons from taking a journey aboard a rocket-

powered transport, this deceleration level is no greater than that experienced just for thrills during a roller-coaster ride at most amusement parks.

Thermodynamic Considerations

Figure 35 compares maximum re-entry temperature of the PEGASUS/GT vehicle with equivalent values for the X-20 Dyna-soar. The 4,000°F nose temperature of the X-20 is analogous to the 1,340°F nose temperature of Pegasus, when the Pegasus nose can be actively cooled by circulation of 2,500 lb. of liquid hydrogen. Initially, this nose temperature was restricted to 600°F, maximum. Since stainless steel coolant tubes can effectively withstand the 1,340°F temperature, a 50% reduction in coolant weight is realized, commensurate with the higher allowable temperature. As an added fringe benefit, the shock wave propagated from the Pegasus blunt entry body will protect the fin leading edges from severe heating conditions. The temperature of the fin leading edge of the X-20 glider is shown

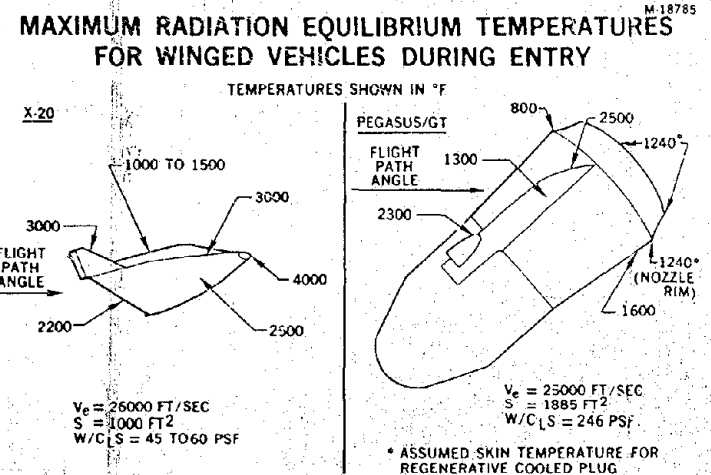


Figure 35

to be approximately 3,000°F; of the Pegasus vehicle, approximately 2,500°F. It should be noted, however, that these values are theoretical maximum temperatures and do not allow for any heat-sink capability of the vehicle structure. Subsequent figures present actual maximums for significant hot spots on the vehicle structure. The Pegasus fins were sized to allow approximately 1,020 sq. ft. of total surface. The entire area can be contained within the envelope defined by the external hydrogen tanks. Hence, wind shear gradients during boost are not imposed directly on the fin surface; otherwise, severe destabilizing moments would result.

Figure 36 is a plot of the fin leading-edge temperature and indicates that a maximum of 2,250°F

PHYSIOLOGICAL DECELERATION COMPARISONS

M-18784

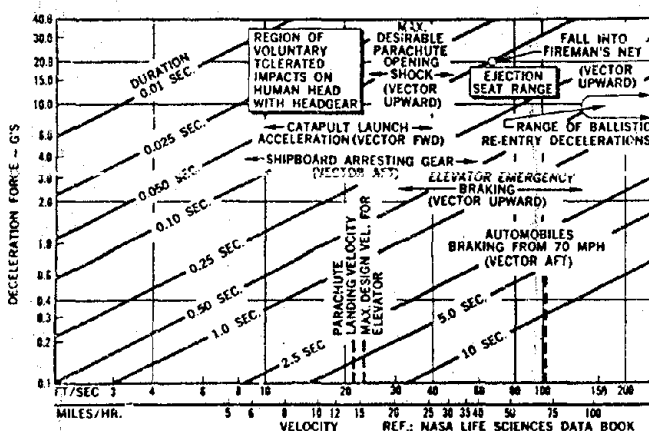


Figure 34

PEGASUS/GT FIN LEADING EDGE TEMPERATURE HISTORY

LEADING EDGE R = 1.5 FT
VE = 25,000 FPS
S = 1885 FT² $\gamma_E = 2.6^\circ$
W/C_LS = 246 PSF $\alpha = 52^\circ$
MATERIAL ASSUMED IS
MOLYBDENUM (.1 IN. THICK)
REF: DYNA-SOAR LEADING
EDGE MAX TEMP = 3000°F

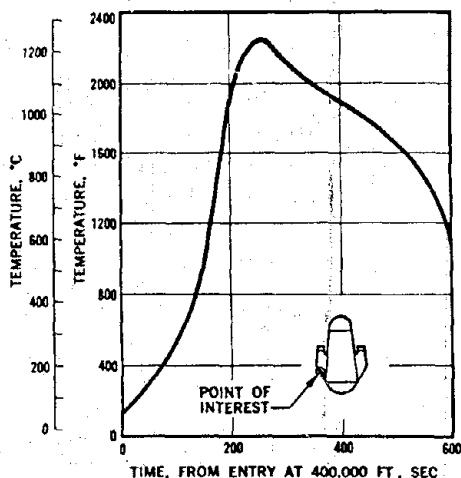


Figure 36

is reached approximately 240 sec. after atmospheric entry (as compared to the 2,500°F theoretical maximum shown in Figure 35 and the 3,000°F leading edge temperature of the X-20). The plot of Figure 36 is based on an assumed leading-edge radius of 1.5 ft. and molybdenum construction with a skin thickness of 0.1 in.

Figure 37 is an historical plot of the main-body entry temperature for Pegasus, indicating that a maximum of approximately 1,160°F is reached about 330 sec. after atmospheric penetration. This temperature is associated with a 3-in. honeycomb sandwich construction for the underside of the body, and a Rene 41 external skin, approximately 0.08-in. thick. When thickness and resulting heat stowage capability are attributed to the material of the vehicle, it is evident that the theoretical maximum of 1,600°F (see Figure 35) will be reduced to approximately 1,160°F. The 1,885-sq.-ft. reference area, indicated in Figure 37, is the cross-sectional area of the 49-ft.-dia. base of the vehicle.

PEGASUS/GT MAIN BODY RE-ENTRY TEMPERATURE HISTORY

VE = 25000 FPS
S = 1885 FT²
W/C_LS = 246 PSF
 $\gamma_E = 2.6^\circ$
 $\alpha = 52^\circ$

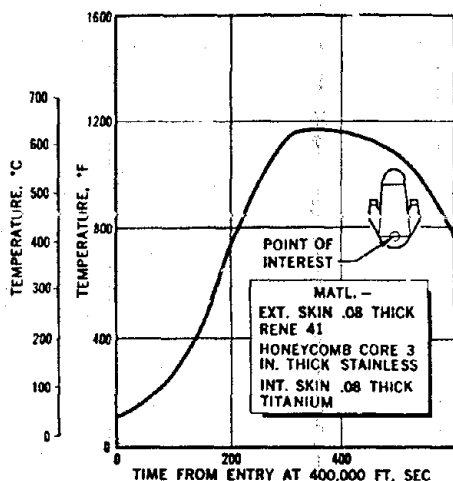


Figure 37

Figure 38 identifies the materials which were selected for the various high-heat-load locations on the Pegasus vehicle during exposure to the entry environment. The upper curve reflects the reduction in the ultimate tensile strength of Rene 41 stainless steel as a function of elevated temperatures. The design point on the curve indicates the maximum temperature to which the Rene 41 vehicle underside will be subjected. At 1,160°F, this material loses only 11% of its room-temperature strength. Correspondingly, the central curve is a plot of the strength decrease (at elevated temperatures) of molybdenum. As indicated, at least 45% of the room-temperature strength remains (at 2,250°F) in the fin leading edge, constructed of molybdenum. The lower plot is indicative of the strength degradation of titanium, the material used for the top surfaces of the fins and centerbody. At the hottest spots on the body (800°F), 60% of the room temperature strength of the material would still remain.

ULTIMATE TENSILE STRENGTH VS ELEVATED TEMPERATURE

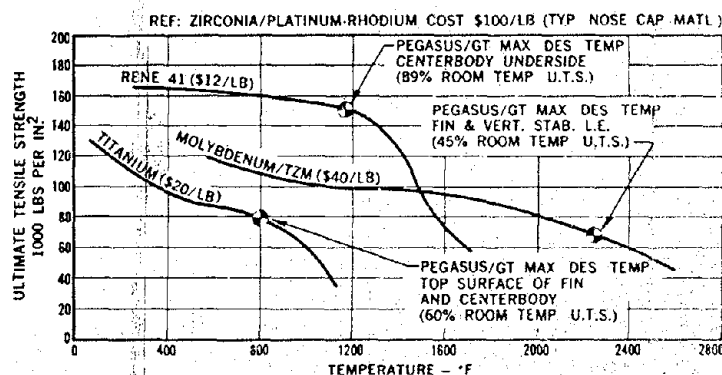


Figure 38

Landing Accuracy

Douglas recently investigated ballistic recovery from an Earth-orbital altitude of 300 n.mi. Figure 39 tabulates the incremental errors assumed for each of the factors contributing to the 3-sigma touchdown dispersions (on land). The largest downrange error increment (40,000 ft.) was attributed to the uncertainty of atmospheric conditions (variations from a standard day), in order to provide the vehicle with all-weather recovery capability. An elliptical area of 62 sq. n.mi., was predicted. This ellipse would have a semimajor axis of about 8 n.mi. and a semiminor axis of about 2.5 n.mi. The 3-sigma dispersion would contain 99.7% of all impact locations. For comparison, a circular error

PREDICTED TOUCHDOWN DISPERSIONS (BALLISTIC RECOVERY FROM 300 N. MI. ORBIT)

M-18789

ERROR SOURCE	ERROR MAGNITUDE (ASSUMED)	3 Σ DISPERSIONS (FT.)	
		DOWNRANGE	CROSSRANGE
DE-ORBIT GUIDANCE	1.1 FT/SEC	20,000	1,000
INITIAL POSITION	1 N.MI.	6,100	2,500
WEIGHT	1%	4,000*	—
C _D	1%	4,000	—
TIMING (RETRO-INITIATION)	0.4 SEC.	12,000	—
ATMOS. DENSITY	10%	0,000	—
WIND DRIFT	99% AMR	10,000	10,000
C.G. OFFSET	3 IN.	10,000	10,000
ROOT SUM SQUARE, TOTAL		48,000 FT* (a = 7.9 N.MI.)	15,000 FT* (b = 2.5 N.MI.)
ELLIPTICAL PREPARED LANDING AREA REQ. RADIUS OF CIRCLE CONT. 50% OF DISP.		62 N. MI. ² CEP = 2.05 N.MI.	

* ASSUMES DUMPING OF RESIDUAL PROPELLANTS
NOTE: NO TERMINAL GUIDANCE CONSIDERED - RECOVERY WITHIN ORBITAL PLANE -
ENTRY ANGLE = 3.5°

Figure 39

probability (CEP), which would contain 50% of all dispersions, was calculated to have a radius of about 2 n.mi. However, this table considers purely ballistic entry--without lifting maneuverability, terminal guidance, or man-in-the-loop to assist in narrowing down the size of the pre-landing area required.

Figure 40 is a plot of the maneuverability which could be incorporated into a low L/D vehicle such as Pegasus. When control is initiated at the beginning of the 10.5-min. atmospheric entry, a downrange and crossrange maneuver capability can be acquired as shown. The nominal design point of the Pegasus vehicle, which allows a maximum deceleration of 3 g's during entry, would be satisfied with a constant bank angle of 50°. Under these conditions, the downrange maneuver capability could be varied by 1,500 n.mi. with the crossrange touchdown point controlled to approximately 120 n.mi. Although a lifting entry vehicle with an L/D greater than 0.425 would provide more control and maneuverability, the L/D of Pegasus appears to suitably correct for the ballistic dispersions shown in Figure 39.

PEGASUS/GT MANEUVERABILITY ENVELOPE

M-18790

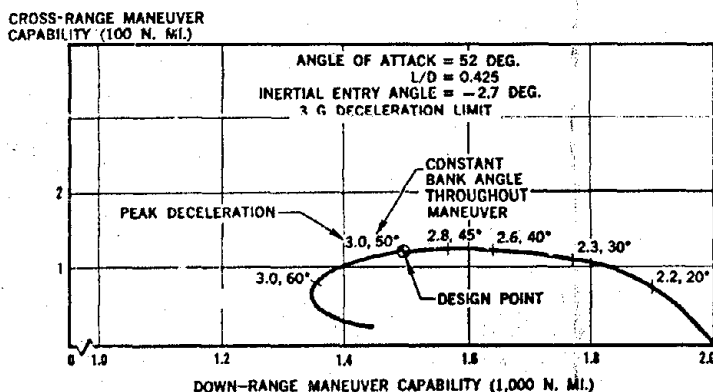


Figure 40

After velocity cancellation by the use of drag techniques, selective segments of the Pegasus propulsion system would be reignited to provide corrective horizontal translation capability for vernier adjustment to the touchdown location. It is estimated that approximately 15,300 lb. of onboard propellant would be required to cancel the 550-fps terminal velocity, and then allow the vehicle to hover for 22 sec., pitch over 10° and translate 1,000 ft. horizontally in any direction. The hover maneuver would consume 17,900 lb. of propellants while resisting gravitational acceleration for 22 sec. (700 fps).

Aerodynamics of Pegasus Hypersonic Flight

Figure 41 identifies the center of gravity (CG) and center of pressure (CP) locations which are necessary to determine the vehicle flight characteristics during glide entry. Perhaps the most significant advantage of entering the atmosphere base-first is the resulting forward location of the CG when the largest concentration of mass (the engine) is located in the nose of the

PEGASUS/GT AERODYNAMIC CONFIGURATION DEFINITION

M-18791

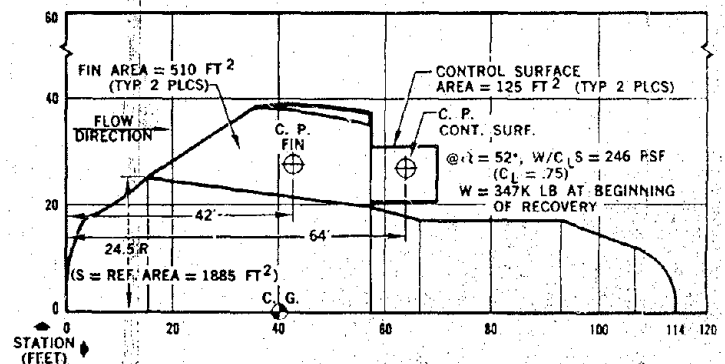


Figure 41

vehicle. Although the liquid hydrogen cooling during entry and the availability of main engines for terminal retrothrust are also attractive benefits, the glide stability offered by the engine-forward condition is the dominant advantage. With the vehicle CG only 2 ft. forward of the fin CP (see Figure 41), the vehicle has positive static stability and requires only a minimal control surface area. By comparison, when a landing engine is incorporated into the aft end of an X-20 type of entry glider, severe control problems result due to the relative location of the CG (aft of the CP).

Figure 42 defines the Pegasus control surface effectiveness on static stability. The control

**PEGASUS
LIFTING ENTRY**
CONTROL SURFACE
EFFECTIVENESS ON
STATIC STABILITY

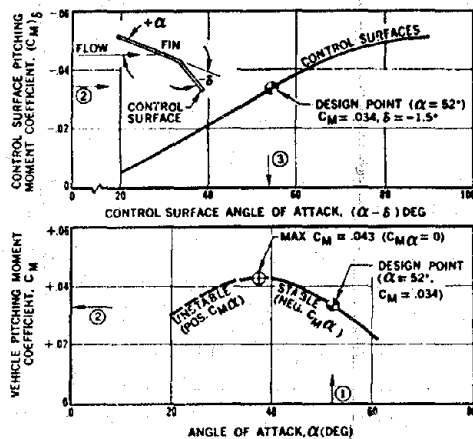


Figure 42

surface pitching moment coefficient (about the CG) is presented in the upper plot as a function of the control surface angle of attack (α), which is apparent to the control surface (flap). The moment coefficients shown are all negative. The lower plot presents the vehicle pitching moment coefficient (without flaps) as a function of α . These coefficients are all positive in the α range which produces reasonable L/D values.

To fly a vehicle without continuous attitude control, the vehicle moment must be zero. To obtain this condition of trim, the vehicle pitching moment found in the lower plot must be corrected by a control surface deflection which produces a moment of equal magnitude (see the upper plot). When a stable and trimmed vehicle encounters a wind gust or other disturbing influence (which rotates the vehicle to some other α), the vehicle will automatically return to the original trim condition and, hence, to steady flight. On the other hand, when an unstable and trimmed vehicle encounters a gust which produces a change in α , the vehicle will depart from the original (desired) α position, unless the flaps are actuated for correction. Obviously, the continuous actuation of control surfaces to maintain the desired α is unattractive.

Under the Pegasus nominal design condition of $\alpha = 52^\circ$, a moment coefficient ($C_M = 0.034$) is induced by the lever arm distance between the CG of the vehicle and the CP of the fins. The upper plot of Figure 42 indicates that a flap deflection angle (δ) of 1.5° downward would trim the vehicle to this desired α position. If an unpredictable gust momentarily displaced the vehicle to $\alpha = 55^\circ$, the vehicle would automatically return to $\alpha = 52^\circ$, without flap actuation for flight correction. This tendency is explained by the fact that when $\alpha = 55^\circ$, the vehicle C_M equals 0.030, as shown. Under this condition,

$\alpha - \delta = 56.5^\circ$, which necessitates a control surface $C_M = 0.036$; hence, the total $C_M = 0.030 - 0.036 = -0.006$, and the vehicle will tend to return to the original $\alpha = 52^\circ$ due to the negative (stable) value of C_M .

The upper plot of Figure 43 presents the flap deflection angles required for trimming the vehicle at various angle of attack (α) values. The $\delta = -1.5^\circ$ value previously discussed is so small that the possibility of reducing the total flap area from 250 sq. ft. to 175 sq. ft. should be investigated. The lower plot shows that when $\alpha = 52^\circ$, this results in the maximum trim lift-to-drag ratio (L/D) = 0.425 (including flap contribution) necessary for optimizing the entry trajectory. The portion of both plots designated by the shaded area ($\alpha < 38^\circ$) corresponds to the positive (unstable) condition of $C_M \alpha > 0$, shown in the lower plot of the previous figure.

**PEGASUS
LIFTING ENTRY**
(TRIM RANGE FOR
ANGLE OF ATTACK)

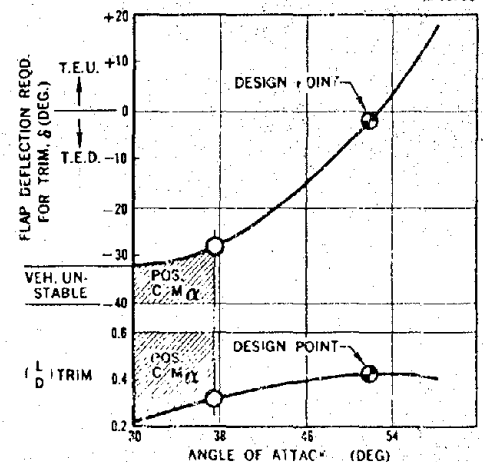


Figure 43

Figure 44 illustrates the effect of α on the lift coefficient (C_L) and the drag coefficient (C_D), for trim conditions. As previously discussed, the α providing the maximum value of L/D was selected as the design point. However, the advantages of flying at maximum C_L ($\alpha > 58^\circ$) should be investigated.

**PEGASUS
LIFTING ENTRY**
(TRIM VALUES OF
 C_L AND C_D)

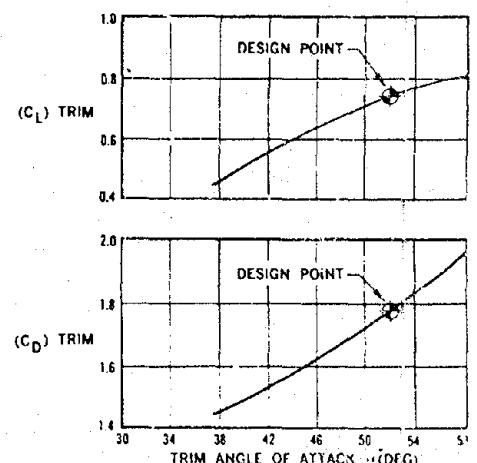


Figure 44

Terminal Landing Phase

Figure 45 illustrates how the pilot and copilot's compartment incorporates unobstructed downward-vision windows for confirming the accuracy of the touchdown location at the spaceport. The Pegasus propulsion system has 16 toroidal combustion chambers, which each produce 263,000 lb. of thrust at liftoff. After re-entry, four engine modules (throttled) are ignited at an altitude of 2,500 ft. to provide retrothrust for terminal velocity cancellation. At reignition, only 610,000 lb. of total thrust are required to produce 2 g's of deceleration. The thrust of these engines is progressively modulated for 12 sec.; they then produce 305,000 lb. of total thrust to balance the recovered weight. Roll control is provided by the attitude control system. Since only four modules (of the available 16) are operated at almost half-thrust during this maneuver, extensive engine redundancy and improved mission reliability are provided at no additional weight penalty. After the hover maneuver, the engines are automatically cut off when the landing legs are compressed. After landing, the vehicle could be moved a short distance by taking-off directly from its landing legs.

PILOT'S DOWNWARD VISION
(PRIOR TO TOUCHDOWN)

M-18795

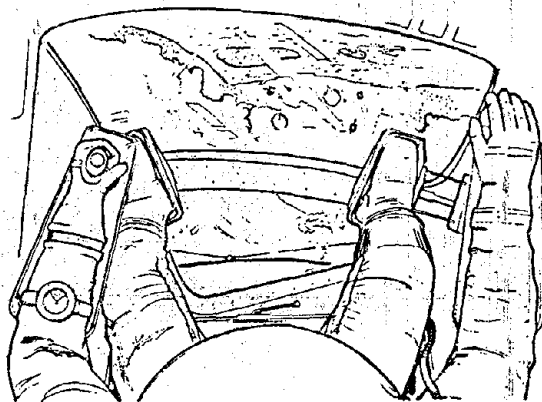


Figure 45

In the event of an emergency landing at a remote, inaccessible, land-locked location, limited propellants could be trucked in on ground vehicles which can traverse the difficult terrain. After refueling, the vehicle could then make a short flight to the nearest seacoast, where a crawler would lift the vehicle and transport it to a waiting barge. Return of the vehicle to the refurbishment building at a coastline spaceport would then be accomplished.

Although the suggestion of launching a booster for even a short hop without a launch pad may

appear rather unrealistic, "it ain't necessarily so." The Apollo mission depends on precisely such an operation for the return phase from the moon. The lunar excursion module (LEM) is launched (while supported by its four legs) from the lunar surface, where no Cape Kennedy is known to exist.

In time, perhaps, the launch complex which was required at the outset of the mission, may be dispensed with although it appears that initial operations would be conducted from a launch pad. Landing maneuvers can be conducted with a greatly reduced thrust level since the vehicle is essentially empty. Therefore, the problems of engine noise and exhaust plume effects on the touchdown surface are significantly diminished.

GROUND OPERATIONS

One major consideration in a PEGASUS/GT rocket transport mission would be the time required to prepare the vehicle for flight-readiness. The time-consuming preparations and countdown procedures required for today's expandable boosters would render a rocket transport mission virtually impractical. Since Pegasus would not be operational until the early 1980's, it is not unrealistic to postulate that after booster reuse has become commonplace, the flight-readiness time may be diminished to a level comparable to that of today's commercial aircraft, that is, from touchdown to the next consecutive flight.

For the PEGASUS/OB, it was estimated that 3.5 months would be initially required for vehicle turnaround time from first launch to first relaunch. This conclusion was based on the following time estimates: (1) 24 days for vehicle refurbishment, (2) 42 days for checkout, (3) 28 days for assembly, and (4) 7 days for vehicle stay time on launch pad. It is impossible to postulate conclusively the degree of increased confidence level in vehicle subsystems and the launch preparation reductions which will result from repeated booster reuse. Only operational experience can be substituted for speculation and conjecture on these vital considerations. It is imperative that time-consuming preflight operations must be minimized; therefore, the Pegasus vehicle would contain onboard automatic checkout equipment to provide instant readiness.

With the premise of reduced turnaround time accepted, the major consideration remaining to influence the feasibility of rocketborne transports would be the time required for loading and un-

loading of VIP passengers and high-priority cargo. Figure 46 tabulates estimated loading times of a Pegasus personnel transport in comparison with an equivalent DC-8 jet airplane. The additional 18 tons of cargo payload which Pegasus carries is assumed to be loaded on 5,000-lb pallets. Cargo loading and unloading would require almost twice as long as for an equivalent number of DC-8F's because the Pegasus cargo compartment is 50 ft. above ground. Assuming that the passenger-loading gantry incorporates elevator platforms to deliver personnel to the four decks (three doors per level) of Pegasus, the rocket transport would require approximately 40% more loading time than a DC-8. During unloading, it is estimated that passenger (and baggage) would require more than twice as long to disembark from the rocket transport.

GLOBAL TRANSPORT VS PLANE LOADING AND UNLOADING TIME COMPARISON

M-18796

EVENT (TIME IN MIN)	170 PASSENGERS + BAGGAGE			
	PEGASUS GLOBAL (1) VEHICLE		AIRCRAFT COMM (1) DC-8	
LOADING TIME:	PASSENGERS	BAGGAGE	PASSENGERS	B. AGE
FROM TERMINAL TO LOADING RAMP	42	25	30	25
LOADING RAMP TO VEHICLE	8	12	8	10
FINAL READINESS	5	3	7	5
TOTAL	55	40	45	40
UNLOADING TIME:				
FROM TOUCHDOWN TO UNLOADING	28	40	8	16
UNLOAD RAMP TO TERMINAL	10	15	7	9
BAGGAGE RE-CLAIM	17	---	10	---
TOTAL	55	55	25	25

Figure 46

Figure 47 compares the Pegasus vehicle, with its landing weight of 305,000 lb., with three typical aircraft, which vary from 270,000 lb. to 80,000 lb. at touchdown. During an emergency landing, the Pegasus vehicle (even when only three of its four legs are loaded on touchdown) can land on any type of unprepared terrain except for quicksand or silt. In contrast, the B-52C and DC-8 could not land on anything other than hard rock, which has an allowable bearing pressure of 50 tons per sq. ft. The C-118, however, could be supported by soft rock, which has an allowable bearing pressure of 8 tons/sq. ft. In case of an emergency, Pegasus could land at sea. Under emergency conditions, the weight of a Pegasus vehicle could readily be supported by coarse sand (such as is found in deserts) if moderately sized landing pads (9 sq. ft.) are used on each of the four landing legs.

GLOBAL TRANSPORT VS AIRPLANE LANDING LOAD COMPARISON

11-18797

SOIL TYPE	ALLOWABLE BEARING PRESSURES TONS/SQ. FT.	PEGASUS GLOBAL TRANSP. 4 LANDING LEG CONFIG /3 LEGS (LOADED) LANDING WT. = 305 K LB FOOT-PRINT DIMENSIONS (LOAD/LEG)	AIRCRAFT COMPARISONS* (AVG. BEARING LOAD TIRE PRESSURE + 10%)		
			B-52C (270K LB L.W.)	DC-8 (200K LB L.W.)	C-119 (80K LB L.W.)
QUICKSAND, SILT	0.5	LOADS WILL EXCEED SOIL BEARING STRENGTH			
DRY CLAY OR FINE SAND	2.5	4' X 5' (2.5 T/FT ²)			
HARD CLAY & SAND (MIXED)	5.0	4' X 3' (4.2 T/FT ²)			
GRAVEL & FIRM COARSE SAND (MIXED)	6.0	3' X 3' (5.5 T/FT ²)			
SOFT ROCK	8.0	2.5' X 2.5' (7.8 T/FT ²)			6.4 T/FT ²
HARD ROCK	50.0	2.5' X 2.5' (7.8 T/FT ²)	18 T/FT ²	14.4 T/FT ²	

*DISREGARDS LENGTH OF PREPARED LANDING STRIP REQUIRED

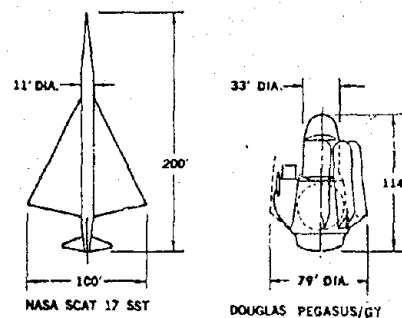
Figure 47

COST CONSIDERATIONS

Figure 48 compares the physical size of Pegasus with a proposed version of the supersonic transport (SST). Although Pegasus has approximately twice the payload capability of the SST, its gross weight is almost seven times greater; yet its landing weight is only 11% higher. However, Pegasus has a range more than twice as great as the SST. Its cruising speed would be 8.5 times that of the SST. Undeniably, one parameter influencing the evolution of a Pegasus-type vehicle will be its cost-effectiveness. As reflected in Figure 48, it does not appear that a rocket-powered vehicle using high-energy cryogenic propellants can effectively compete with airplanes on an operational cost basis.

TRANSPORT VEHICLE COMPARISON

M-18798



TOTAL USEFUL PAYLOAD WEIGHT	33,200 LBS	73,400 LBS
PASSENGERS	26,200 LBS (125 PASS. + BAGGAGE)	37,400 LBS (170 PASS. + BAGGAGE)
CARGO	7,000 LBS	36,000 LBS
GROSS WEIGHT	500,000 LBS	3,340,000 LBS
LANDING WEIGHT	275,000 LBS	305,000 LBS
THRUST	160,000 LBS	4,200,000 LBS
RANGE	4,000 ST. MI.	8,750 ST. MI.
CRUISING SPEED	2,000 MPH	17,000 MPH
TRANSIT TIME FOR 7600 N.MI. DISTANCE	4.4 HRS (THEO.)	39 MIN
RUNWAY LENGTH - TAKEOFF	9500 FT	0
SEAT MILE COSTS (D.O.C.)	1 1/2 CENTS/ST. MI.	UNDETERMINED
SEAT MILE COST (PROD. ONLY)	1/2 CENTS/ST. MI.	6 1/2 CENTS/ST. MI. (BASED ON 260 PASS.)

Figure 48

When a large number of vehicle reuses can be realized, the vehicle cost per flight is reduced to an insignificant level, and the operational costs are principally comprised of launch costs and propellant costs. Historically, airplane fuel costs constitute about 50% of the total direct operating cost. Figure 49 presents preliminary fuel cost estimates for transport missions. The total cost for the combined rocket propellants, at a mixture ratio of 7 to 1, is approximately 5¢/lb. as compared to 2¢/lb. for kerosene consumed by the aircraft engine. In terms of cost per seat-mile, considering only the fuel, Pegasus would cost 8.7 times as much to operate as a conventional jet aircraft.

Although reduced cost of cryogenic propellants can be predicted with mass production, they cannot conceivably be produced at a cost level competitive with JP-4 fuels, unless some unforeseen breakthrough in production technology can be accomplished. Moreover, a rocket (which must carry onboard oxidizer) would consume far more propellant weight for the same range and payload than a conventional air-breathing aircraft. (See Figure 49.) For example, fuel for the three representative aircraft shown in the table, costs an average of 3.7¢/ton-mile. The cost of Pegasus fuel per ton-mile would be 12.5 times as great. Although a cost analysis has not been conducted on the PEGASUS/GT, its antecedent orbital booster version has been esti-

GLOBAL TRANSPORT VS TURBINE AIRCRAFT FUEL COST COMPARISON

M-187

	PEGASUS/GT (LO ₂ /LH ₂ 7/1)	AIRCRAFT (JP-4 FUEL)***		
		DOUGLAS DC-8	CONVAIR 440	LOCKHEED ELECTRA
USEFUL PAYLOAD (TONS)*	36.7	16.22	11.72	9.45
RANGE - ST. MI. (MAX)	8,750	6,000	3,750	3,400
NO. OF REFUELING STOPS FOR 8,750 ST. MI. TRIP	0	1	2	2
FUEL CONSUMPTION (LBS/MI)*	332.0	33.2	28.7	17.2
PASSENGER CAPACITY*	260**	123.8	93.3	75.1
AVERAGE SPEED (ST. MI./HR)*	17,000	485	466	299
SPECIFIC FUEL COST (¢/TON-MILE)	46.3	3.56	4.27	3.18
SPECIFIC FUEL COST (¢/SEAT-MILE)	6.55	0.467	0.536	0.40
FUEL COST (¢/LB)	5.12	1.74	1.74	1.74

ASSUMED PROPELLANT COSTS: JP-4 AT \$0.113/GAL; LO₂ AT \$0.02/LB, LH₂ AT \$0.27/LB

* AVERAGE ALL COMMERCIAL CARRIERS - 1963 STATISTICS

** 18 TONS OF CARGO CONVERTED TO EQUIVALENT NUMBER OF PASSENGERS
(90 AT 400 LBS/PERSON; INCL CABIN, PRESS. SYST., SEATS, ETC.)

*** FUEL COST IS APPROXIMATELY 50% OF TOTAL D.O.C.

Figure 49

ated to cost \$2.75 billion for development. The PEGASUS/OB vehicle would cost an average of \$44 million per copy, assuming a fleet of 30 vehicles. The first flight vehicle is estimated to cost \$76 million, which includes \$13 million for the two-man crew compartment (based on Gemini costs).

Clearly, the predominant argument for development of a rocket-powered transport is not one of increased economy over the air-breathing jet. Conversely, it must be recognized that the conventional jet aircraft cannot deliver passengers and cargo to orbit.

CONCLUSION

In conclusion, it must be re-emphasized that development of the PEGASUS/OB orbital booster version is a mandatory prerequisite to the evolution of its global transport counterpart. Only in such an event, would VIP rocket transport prove feasible. Both systems depend in large measure on the same advanced propulsion technology and would require enormous expenditures of time and funds for extensive preliminary research. However, the attendant reductions in vehicle size and weight offered by such sophisticated propulsion systems appear to justify their development.

Unfortunately, the Pegasus study failed to identify any cost-reduction advantage associated with the global transport derivative. In the absence of this urgent economic stimulus, the paramount justification for its potential desirability is the dramatic increase in speed which it offers.

Perhaps the most valid reason for man's insatiable desire for ever-increasing speed is not to provide him with instant transportation but with instant communication. Each year privately-owned business aircraft in the United States fly at least a million hours more than the 1,800 aircraft in our fleet of commercial airliners. Meanwhile, American air carriers fly 72 million people over 53 billion passenger miles each year, and it may be assumed that a certain proportion of these, too, are moving from place to place for business reasons.

Studies have shown that most business travelers fall into the following categories: professionals, owners and managers, salesmen, and skilled technicians. They may be diplomats, engineers, scientists, or specialists in construction, agriculture, manufacturing, sales and distribution, and government.

Tens of thousands of such people fly regularly to be present at more places in a given period of time--to consult, confer, inspect, decide, and lead in a manner impossible to accomplish from a distance, even with television. Is it reasonable to suppose that after another 20 years of accelerated accumulation of knowledge in science, professions, and business, it will be possible to convey specialized wisdom and critical solutions by any means other than instant personal contact? A time will come when meaningful interviews will only be possible by direct face-to-face encounter. It appears that the voluminous knowhow which we are generating soon will get so complex that it must always be shipped in the only feasible container--the human brain.

Now that the Space Age has provided us with the means to travel at tremendous velocities, can we identify any real need for this mode of transportation? When this new medium is recognized as a means of communication rather than transportation, there will be no limit to the speed desired; and, as that speed is needed, it will be made available.

Perceptive observers have noted that historically there has appeared some new material or technique which has been a stimulus to human progress. Fire, writing, iron, the wheel, the sailboat, steam power, electricity, dynamite, radio, and nuclear fission have come in a sequence which, regardless of incidental military employment, closely parallels the progress of civilization. Accordingly, the world's need for rapid extension of the communication network appears as clear as was the need for fire to equip man for the Ice Age or writing to pass along his experiences.

History assures us that there is much more of the drama of man's achievements yet to unfold--but in what direction?

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