STAR-RAKER
AN AIRBREATHER/ROCKET-POWERED, HORIZONTAL TAKEOFF
TRIDELTA FLYING WING, SINGLE-STAGE-TO-ORBIT
TRANSPORTATION SYSTEM

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Abstract

Star-Raker is a versatile space transportation system (STS) that eliminates the need for expensive launch facilities, vehicle assembly buildings, and expendable boosters. It is a system that can operate from any large airport equipped with cryogenic facilities, can carry a 100-ton payload into any 556-km (300 nm) orbit, uses multicycle airbreather propulsion, and is lightweight. These qualities contribute to operational flexibility, and can play key roles in fulfilling future earth/space transportation requirements.

Introduction

Studies conducted by NASA and industry[1,2] have clearly established the need for a new earth launch vehicle (ELV) if the satellite power systems (SPS’s) are to be economically viable. Typically, the ELV’s conceived to meet this need are vertically launched and fully recoverable. To be cost-effective, they must transport very large payloads (100,000 kg or more) to orbit. These concepts, although based on acceptable technology, introduce uncertainties related to major program planning and operational usage.

Development of large ELV’s can be justified only in the context of full SPS production go-ahead. This is dictated by the substantial funding required for development of vehicles and launch facilities whose projected applicability is limited solely to the demands of an SPS program. A potential funding dilemma is posed if a near full-scale prototype SPS at geosynchronous orbit is required for proof of concept, inasmuch as a new ELV could save billions of dollars in launch operations costs if it were available. What is needed, therefore, is an ELV concept that offers the potential of low dollars per pound to orbit and is also applicable to future NASA and Air Force programs with considerably lesser demands than SPS. Thus, the economic risk of early development would be essentially negated.

Operationally, the concept of very large, vertically launched, all-rocket, boosters poses formidable environmental and logistical concerns. Atmospheric and launch acoustics pollution will be high for the multiple-launches-per-day requirements. Water recovery of expended ballistic stages and subsequent refurbishment, stacking, payload integration, and countdowns with short turnaround schedules at low cost represent a solution full of credibility issues for launch systems recovery. There is a need, therefore, for a new ELV concept which can counter many of these objections.

While conducting the Satellite Power System (SPS) Concept Definition Study (NAS8-32161), Rockwell introduced a horizontal takeoff, tridelta flying wing configuration spacecraft, identified “Star-Raker.” The spacecraft is powered with turbofan, air turboexchanger, ramjet multicycle airbreather engines, and rocket engines. The configuration, weights, and performance estimates for this vehicle were based on 1968 conceptual efforts prior to the Space Shuttle procurement. Justification for introducing the concept into the recently completed SPS study was founded on ROM cost correlations and the obvious operational advantages of such a system if it is proved feasible.

Previous investigations at Rockwell,7 such as 1958 Navaho follow-on applications, Dyna-Soar, and initial recoverable orbital carrier (ROC) studies, included turbojet/ramjet propulsion limited to 10,660 m (35,000 ft) altitude and Mach 3 velocity. The studies included winged vehicles; however, the wings were not permitted to contain cryogenic propellants. The resultant maneuver from Mach 3 low-gamma (flight path angle), high-specific impulse (Isp) flight inherent in air propulsion to the high-gamma, low Isp flight inherent in rocket propulsion negated most of the high Isp potential of air propulsion.

Later, recoverable orbital carrier studies increased airbreather propulsion up to Mach 6, using SWAT-201 high-bypass supersonic turbofan/ramjets. However, as in previous studies, wet designs were not permitted. The results of such designs indicated airbreather propulsion was of significant advantage for two-stage launch systems and marginal for single-stage launch systems.

“A-BALL” studies in 1966 and 1969 permitted wet wing design of “conventional” wing/body systems. Center-of-aerodynamic-pressure (c.p) excursion across all Mach ranges proved to be excessive. However, the potential of single-stage-to-orbit launch systems was reasonably established. A review of supersonic transport (SST) studies conducted by Rockwell, Boeing, and Lockheed produced approximately 15 candidate planform systems with low c.p. excursions. The tridelta used in the Star-Raker design had the best c.p. versus Mach number behavior, and was chosen for further study in 1970.

The results of the 1970 Rockwell studies and the SPS transportation requirements from a NASA contract in 1976 were used as inputs for the 1977 IRDA studies. Data generated in 1977 provided a technology base for ongoing 1978 study effort. This paper presents an overview of a reference vehicle, including system descriptions and data summaries, for IRDA studies ending in 1978.

Operational Features

The Star-Raker concept adapts existing and advanced commercial and/or military air transport system concepts, operations methods, maintenance procedures, and cargo handling equipment to include a space-related environment. The principal operational objective is to provide economical, reliable transportation of large quantities of material between earth and low earth orbit (LEO) at high flight frequencies with routine logistics operations and minimal environmental impact. An associated operational objective is to reduce the number of operations required to transport material and equipment from their place of manufacture on earth to low earth orbit. Goals for the Star-Raker concept, which are in addition to the quantitative requirements for logistic support of the SPS, include both NASA and Air Force objectives and requirements, such as the following.
- Total vehicle reusability—with many reuses
- Rapid turnaround
- Ferry capability with cargo between airfields
- Minimized launch costs
- High reliability of delivery
- Ability to reach any LEO plane from alternate launch sites (KSC, VAFB, and others) and return to the same site, including single-pass orbits
- Cargo security

Operational features derived in the study include:

- Single orbit up/down, from/to the same launch site (at any launch azimuth subject to payload/launch azimuth match)
- Capability of attaining a 556 km (300 nmi) equatorial orbit when launched from KSC
- Takeoff and landing on 2440- to 4270-m (8000- to 14,000-ft) runways (launch velocity ≈ 225 knots; landing velocity ≤ 115 knots)
- Simultaneous multiple launch
- Total system recovery, including jettison and recovery of takeoff gear at the launch site
- Payload cost potential of $22 to $33 per kilogram ($10 to $15 per pound) in 556-km (300-nmi) orbits
- Negligible launch site refurbishment
- Powered landing from orbital reentry at any commercial or military airport capable of 747 or C-5A operations
- Aerodynamic flight capability from the payload manufacturing site to the launch site, addition of launch gear and fueling, and launch into earth orbit
- Amenity to alternative launch and landing sites
- Incorporation of Air Force (C-5A Galaxy) and commercial (747 cargo) payload handling, including railroad, truck, and cargo/ship containerization concepts, modified to meet space environment requirements
- Swing-nose loading and unloading, permitting the use of a normal aircraft loading-door facility
- Propulsion system service using existing support equipment on runway aprons or near service hangars
- In-flight refueling options (options not included in reference vehicle data)

Fig. 1 shows a Star-Raker vehicle in an airport alongside a Boeing 747 passenger plane. Multiple launch capability (shown in Fig. 2) combined with orbital operations (Fig. 3), and powered landing at commercial airports (shown in Fig. 4) illustrate the versatile attributes of flight for earth/space missions. Cargo loading, exemplified in Fig. 5, illustrates not only space operations but suborbital point-to-point earth operations that accrue as fall-out from the basic mission capabilities.

Design Features

The Star-Raker utilizes a tridelta flying wing, consisting of a multi-cell pressure vessel of tapered, intersecting cones. The tridelta planform and a Whitecomb airfoil section offer an efficient aerodynamic shape from a performance standpoint and high propellant volumetric efficiency. The outer panels of the wing and vent system lines in the wing's leading edge provide the gaseous
support, power (fuel cell), and other subsystems, including spare life support and emergency recovery equipment.

Ten high-bypass, supersonic-turbofan/air-turbo-exchanger/ramjet airbreather engines with a combined static thrust of 6,277,510 N (1,400,000 lb) are mounted under the wings. The inlets are variable-area retractable ramps that also close and fair the bottom into a smooth surface during rocket powered flight and for either Sanger skip glider or high angle-of-attack initial reentry with atmospheric maneuverable descent.

Fig. 7 shows an inboard profile of the vehicle, illustrating the details of body construction, crew compartment, cargo bay length, LH₂ tank configuration, and location of the rocket engines at rear of fuselage. The hinging and rotation of the nose section for loading and unloading the payloads are illustrated, with indication of view angle from the rear of the nose section during these operations. The multiple landing gear concept shows the position of the nose gear bogie, the jettisonable takeoff gear, and the main landing gear for powered landing.

Fig. 8 presents front and rear views of the vehicle, showing the blended wing, engine inlet ducts, landing gear arrangement, and vertical stabilizer. Also shown are typical sections through the vehicle at the following places:

- The hinge line section (B-B) aft of the crew compartment and forward of the nose gear. Cross-sectional dimensions of the cargo bay are indicated.
- The 40% chord line fuselage section (C-C) illustrating the wing and fuselage construction and the profile of the wing/fuselage fairing.
- The main landing gear station (D-D) illustrating the gear retraction geometry, and the relationship of the gear to the engine air inlet ducts, and the wing construction and profile to the fuselage shape.
The multi-cell wing tank shown in Fig. 9 and enlarged construction details of Fig. 10 show how the aerodynamic lift, propellant pressure vessel, and thermal protection system requirements are integrated into a single, lightweight design concept.

Fig. 10 presents details of the basic multicell structure of the Star-Raker wing. The upper portion illustrates the application of Shuttle-type RSI tile thermal protection system (TPS), and the lower portion shows a potential utilization of a “metallic” TPS.

The wing is an integrated structural system consisting of an inner multi-cell pressure vessel, a foam-filled structural core, an inner facing sheet, a perforated structural honeycomb core, and an outer facing sheet. The inner multi-cell pressure vessel arched shell and webs are configured to resist pressure. The pressure vessel and the two facing sheets, which are structurally interconnected with phenolic-impregnated, glass-fiber, honeycomb core, resist wing spanwise and chordwise bending moments. Cell webs react wing-lift shear forces. Torsion is reacted by the pressure vessel and the facing sheets as a multi-box wing structure.

The outer honeycomb core is perforated and partitioned to provide a controlled passage for purge and gas leak detection as well as a structural interconnect of the inner and outer facing sheets. The construction of the wing structure utilizes the “Inflation Assembly Technique” developed by Rockwell for the Saturn II booster common bulkhead.

Multi-Cycle Airbreather Engine System

Takeoff and climb to 30,400-km (100,000-ft) altitude and 1768 m/s (5800 ft/s) velocity are accomplished by airbreather propulsion. Parallel burn of airbreather and rocket propulsion occurs between 1768 to 2195 m/s (5800 to 7200 ft/s). Rocket power propels Star-Raker from 2195 m/s (7200 ft/s) into orbit.

The multi-cycle airbreathing engine system concept (Fig. 11) is derived from the General Electric CJ805 aircraft engine, the Pratt and Whitney SWAT 201 supersonic wraparound turbofan/ramjet engine, the Aerojet Air Turborocket, the Marquart variable plug-nozzle, ramjet engine technology, and the Rocketdyne tubular-cooled, high T, rocket engine technology.

The multi-mode power cycles include an aft-fan, turbofan cycle; an LH2 regenerative Rankine, air-turbo-exchanger cycle; and a ramjet cycle that can also be used as a full flow (turbojet core and fan bypass flow), thrust-augmented turbofan cycle. These four thermal cycles may receive fuel in any combination, permitting high engine performance over a flight profile from sea-level takeoff to Mach 6 at 30,400-km (100,000-ft) altitude.

The engine air inlet and duct system (Fig. 11) is based on a five-ramp variable inlet system with actuators to provide ramp movement from fully closed (upper right figure) for rocket-powered and reentry flight, to fully open (lower right figure) for takeoff operation.

The engine components include a rotary vane assembly to close off the compressor-turbine assembly at higher Mach numbers. LH2 fuel permits the use of a Rankine-cycle air turbo-exchanger cycle to provide power for the bypass fan. This allows elimination of approximately one-half of the turbofan compressor stages normally needed for fan drive. Heating of the LH2 in outer walls and nozzle plug of tubular construction, in addition to providing fan-drive power, permits stoichiometric combustion in the augmentor/ramjet by cooling of exposed surfaces. The 3058K combustion temperature provides high cycle efficiency.
The study scope did not permit a detailed evaluation of engine components to provide further, more accurate calculation of the performance capability of this engine concept. Engine manufacturers are best equipped to further refine the design and provide real data on concept feasibility and system weight.

The inlet area was determined by the engine airflow required at the Mach 6 design point. The configuration required 6,227,510 N (1.4x10^9 lb) of thrust at the Mach 6 condition and at least 5,337,870 N (1.2x10^9 lb) of thrust for takeoff. This resulted in an inlet area of approximately 111 m^2 (1200 ft^2), or 11.1 m^2 (120 ft^2) per engine for a 10-engine configuration. To provide pressure recovery with minimum spillage drag over the wide range of Mach numbers, a variable multi-ramp inlet is required. Inlet pressure recovery efficiency versus velocity is plotted on Fig. 12. Higher recoveries are possible for the Star-Raker than for military aircraft, which must operate during more violent maneuvers. However, the pressure recovery must still provide a margin which prevents inlet instability and possible engine flameout from expulsion of the normal shock during transients.

Estimated engine thrust (total of 10 engines) versus velocity is plotted on Fig. 13. Initially, a constant thrust of 6,227,510 N (1,400,000 lb) was assumed for the Rockwell modified Rutowski energy method trajectory analysis (dashed curve of Fig. 13). A tentative airbreather engine performance map was estimated from engine data sources previously described. The engine thrust versus Mach number estimate is shown by the upper solid curve of Fig. 13.

- Supersonic range: Reduction of $I_{SP}$ from 4000 sec at 366 m/s (1200 ft/s) to 3500 sec at 1707 m/s (5600 ft/s) (AB)
- Rocket: $I_{SP} = 455$ sec

**Aerodynamic Characteristics**

The wing shape is a supercritical Whitcomb airfoil with a relatively blunt leading edge, flat upper surfaces, and cambered trailing edges. The trailing-edge camber and the tridelta shape minimize translation of the center of pressure throughout the flight Mach number regime. The blunt leading edge offers good subsonic characteristics, but produces relatively high supersonic wave drag; therefore, further shape refinements are recommended. The wing has a spanwise thickness distribution of 10% at the root, 6% near midspan, and 5% at the tip, providing a large interior volume for storage of LH2 and LO2 propellants.

Subsonic and supersonic aerodynamic coefficients ($C_L$, $C_D$, $c_p$) were calculated using the Flexible Unified Distributed Panel program. Aerodynamic coefficients computed at $M_a = 5.0$ were frozen and used for hypersonic application. Viscous drag due to the skin friction was added in a separate analysis. The resulting aerodynamic coefficients are plotted versus flight Mach number on Fig. 14.

Maximum lift/drag and corresponding lift coefficients and angle of attack versus Mach number are given on Fig. 15.

**Flight Mechanics**

Most of the ascent performance analyses for the Star-Raker vehicle concept were accomplished by employing the lifting ascent program which is based on a modified Rutowski energy method. A second computer program, the Two-Dimensional Trajectory Program (TDTP), was then used to compute the ascent trajectory timeline.

An end-to-end simulation of the Star-Raker (i.e., airbreather horizontal takeoff, climb, cruise, turn, airbreather ascent, rocket ascent, coast and final orbit insertion) with flight optimization including aerodynamic effects was verified by the Langley...
Program to Optimize Simulated Trajectory (POST), developed by Martin-Marietta.

The Star-Raker uses aircraft-type flight from airport takeoff to approximately Mach 6, with a parallel burn transition of airbreather and rocket engines from Mach 6 to 7.2, and rocket-only burn from Mach 7.2 to orbit. Fig. 16 illustrates a nominal trajectory from KSC to 556-km (300-nmi) earth equatorial orbit. Prime elements of the trajectory include:

- Runway takeoff under high-pass turbofan/air turbo exchanger (ATE) ramjet power, with the ramjets acting as supercharged afterburners
- Jettison and parachute recovery of landing gear used only for launch
- Climb to optimum cruise altitude with turbofan and ATE power
- Cruise at optimum altitude, Mach number, and direction vector to earth's equatorial plane, using turbofan power
- Execute a large radius turn into the equatorial plane with turbofan power
- Climb subsonically at optimum climb angle and velocity to an optimum altitude by using high bypass turbofan/ATE/ramjet (supercharged afterburner) power
- Perform an optimum pitch-over into a nearly constant-energy (shallow gamma-angle) dive, if necessary, and accelerate through the transonic region to approximately Mach 1.2, by using turbofan/ATE/ramjet (supercharged afterburner) power
- Execute a long-radius optimum pitch-up to an optimum supersonic climb flight path, by using turbofan/ATE/ramjet power
- Climb to approximately 29 km (95 kft) altitude, and 1890 m/s (6200 ft/s) velocity, at optimum flight path angle and velocity, using proportional fuel flow throttling from turbofan/ATE/ramjet, or full ramjet, as required to maximize total energy acquired per unit mass of fuel consumed as a function of velocity and altitude
- Ignite rocket engines to full required thrust level at 1890 m/s (6200 ft/s) and parallel burn to 2195 m/s (7200 ft/s)
- Shut down airbreather engines while closing airbreather inlet ramps
- Continue rocket power at full thrust

- Insert into an equatorial elliptical orbit 91 x 556 km (50 x 300 nmi) along an optimum lift/drag/thrust flight profile
- Shut down rocket engines and execute a Hohmann transfer to 556 km (300 nmi)
- Circularize Hohmann transfer

The reentry trajectory (shown in Fig. 17) is characterized by low gamma (flight path angle), high alpha (angle of attack) initial atmospheric entry, and aero maneuver descent. The main reentry trajectory elements are:

- Perform delta velocity (ΔV) maneuver and insert into an elliptical orbit 91 x 556 km (50 x 300 nmi)
- Perform a low-gamma, high-alpha deceleration to approximately Mach 20 at an altitude of 82 km (270,000 ft)
- Reduce angle of attack to maximum lift/drag (L/D) for high-velocity glide and crossrange by angle of attack and bank modulation maneuvers to subsonic velocity (approximately Mach 0.85)
- Open inlets and start some airbreather engines
- Perform powered flight to the landing field, land on the runway, and taxi to the dock

Flyback fuel requirements include approximately 300 nmi subsonic cruise and two landing approach maneuvers (first approach waveoff with fly-around second approach).

Required takeoff distances for the Star-Raker were compared with existing data. The distance computed for a tire-rolling coefficient μ = 0.10 agrees with the existing data. Throw weight sensitivity to tire-rolling coefficient is 317 kg (700 lb) (reduction) from μ = 0.02 to 0.20. The data of Fig. 18 show that from 2438 to 3658 m (8000 to 12,000 ft) of takeoff run is reasonable for the current wing-loading, power-loading combination.

The airbreather cruise mode results in an economical orbit plane change from the launch site to the equatorial orbit. Although subsonic cruise takes longer than supersonic cruise, the amount of fuel consumed is substantially less when the orbital plane change is accomplished with subsonic cruise at maximum L/D rather than with supersonic cruise.

Typical ascent trajectory parameters without the cruise leg are shown in Fig. 19. The acceleration load remains less than 2.3 g (76% of Shuttle maximum acceleration), which is a comfortable level for both crew and payload. Maximum dynamic pressure is 60 kPa, which occurs during the airbreather acceleration and is within load limits. From takeoff to burnout, the ascent profile is quite shallow—with flight path angle ranging between -0.7 and 4.5 degrees. Due to the lower AB thrust loading, approximately 75% of flight time is expended during the AB climb/acceleration.
A major advantage of the Star-Raker flying-wing vehicle over the VTO ballistic space vehicle is the efficient use of aerodynamic lift during ascent. Since the ballistic system must overcome large gravity losses during lift-off and the ascent period, it requires an initial thrust/weight \( \geq 1.0 \). However, the Star-Raker can have an initial \( T/W \leq 1 \) because of the efficient aerodynamic wing with high L/D. This high L/D winged vehicle maintains a shallow flight path angle \( (\gamma < 2^\circ) \) during ascent and permits a greater portion of propulsive energy to be expended in kinetic energy of acceleration and velocity rather than in gain in potential energy of altitude. This use of energy is more efficient than that of a vertically launched rocket. Furthermore, the additional lift due to centrifugal force becomes available for flight speeds greater than 300 m/s (10,000 ft/s).

Velocity and altitude histories during vehicle ascent vary for different modes of trajectory shaping. However, the ambiguity of performance comparison between different flight trajectories can be eliminated by comparing a plot which shows performance data versus total energy velocity \( (V^* = \sqrt{V^2 + 2gh}) \), which accounts for potential and kinetic energy variations.

Typical HTO/SSTO performance comparisons of the AB/rocket propulsion and the all-rocket propulsion systems are shown in Figure 20. The difference in throw weight (approximately 186,000 kg \( (\Delta W/S = 49 \text{ kg/m}^2) \)) is the cumulative effect of AB engine operation, as compared to the rocket, in the speed regime less than \( V^* = 2000 \text{ m/s} \). Therefore, an net payload increase will result by going from the all-rocket to the AB/rocket system if the additional AB engine system weight requirement is less than the increase in throw weight.

The mass ratio \( \text{GLOW} / \text{throw weight} \) is approximately 5, 8, and 20 for the AB/rocket HTO, the all-rocket HTO, and the Space Shuttle VTO, respectively. Hence, a well-designed HTO system can improve the throw weight mass ratio over the VTO system.

Weight in orbit is summarized in Table 1. The data enters identified by an asterisk are revised reference vehicle data resulting from Rockwell and NASA/MSFC data exchange in May 1978. Calculations reflect additional fuel reserves, performance losses, and a 10-percent growth factor. Inert weight in orbit was increased from 314,970 kg (694,610 lb) to 352,213 kg (775,800 lb), and airbreather engine thrust of \( 6.227 \times 10^6 \text{ N} (1.4 \times 10^6 \text{ lb}) \) constant was revised to reflect increase in the airbreather thrust potential shown in Figure 13.

The lower payload into the equatorial orbit resulted from the expenditure of cruise fuel from the KSC to the equator. The equatorial orbit payload can exceed 90,000 kg if an equatorial based launch site is used. In this case, the Star-Raker without LOx weighs less than 707,000 kg (1,300,000 lb) and can fly from the continental United States to the launch site with substantial savings in the cruise fuel.

Ascent and descent trajectories of the Star-Raker and the Space Shuttle missions are compared in Fig. 21. Because the performance of airbreathing engines and aerodynamic lifting of winged vehicle depend on the high dynamic pressure, the Star-Raker flies at much lower altitude during the powered climb.

<table>
<thead>
<tr>
<th>Orbit</th>
<th>GLOW ( W_0 \times 10^6 \text{ kg} )</th>
<th>In Orbit ( W_0 \times 10^6 \text{ lb} )</th>
<th>P.L. ( \times 10^3 \text{ kg} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>Equatorial</td>
<td>2.27</td>
<td>409.5</td>
<td>57.2</td>
</tr>
<tr>
<td>cruise from KSC</td>
<td>(5.0)</td>
<td>(902.0)</td>
<td>(126.2)</td>
</tr>
<tr>
<td>28.5° inclined</td>
<td>2.27</td>
<td>441.5</td>
<td>89.2</td>
</tr>
<tr>
<td>orbit from KSC</td>
<td>(5.0)</td>
<td>(972.4)</td>
<td>(196.6)</td>
</tr>
</tbody>
</table>

- Launch from KSC
- Data for 556.0 km (300 nmi) orbital insertion
- Reference wing area \( (S_W) = 3814 \text{ m}^2 (40,900 \text{ ft}^2) \)
- Weight in orbit (excluding payload) = 352,213 kg (775,800 lb)*
- Airbreather
  - Thrust = \( 6.227 \times 10^6 \text{ N} (1.4 \times 10^6 \text{ lb}) \)
  - \( I_{sp} \) = variable
  - Velocity = \( 0 \leq V \leq 1891 \text{ m/s} (6200 \text{ ft/sec}) \)
- Rocket
  - Thrust = \( 14.23 \times 10^6 \text{ N} (3.2 \times 10^6 \text{ lb}) \)
  - \( I_{sp} \) = see chart
  - Velocity = \( 1891 \text{ m/s} (6200 \text{ ft/sec}) \leq V \leq V_{\text{orbit}} \)

*\( W_0 \) = total weight; P.L. = payload
than the vertical ascent trajectory of the Space Shuttle for a given flight velocity. Light wing loading of the Star-Raker contributes to the rapid deceleration during deorbit.

The total enthalpy flux histories, which indicate the severity of expected aerodynamic heating, are shown in Fig. 22. As expected, the aerodynamic heating of ascent trajectory may determine the Star-Raker TPS requirement. The maximum total enthalpy flux of $6.89 \times 10^6$ W/m$^2$ (6000 Btu/ft$^2$-sec) is estimated near the end of the airbreather power climb trajectory. Except in the vicinity of vehicle nose, wing leading edge, or structural protuberances where interference heating may exist, most of the ascent heating is from the frictional flow heating on the relatively smooth flat surfaces.

The descent heating is mainly produced by the compressive flow on the vehicle windward surface during the high angle-of-attack reentry, and is expected to be considerably lower than the Space Shuttle reentry heating.

Aerodynamic and Structural Heating

Preliminary aerodynamic heating evaluations of the Star-Raker configuration were performed for several wing-spanwise stations and the vehicle centerline.

For the wing lower surfaces, heating rates were computed, including the chordwise variation of local flow properties. Effects of leading edge shock and angle of attack were included in the local flow property evaluation. Leading edge stagnation heating rates were based on the flow conditions normal to the leading edge, neglecting cross-flow effects. All computations were performed using ideal gas thermodynamic properties.

Wing upper-surface heating rates were computed using freestream flow properties, i.e., neglecting chordwise variations of flow properties. Heating rates were computed for several prescribed wall temperatures as well as the radiation equilibrium wall temperature condition. Transition from laminar to turbulent flow was taken into account in the computations. Wing/body and inlet interference heating effects were not included in this preliminary analysis. The analysis was limited to the ascent trajectory, since the descent trajectory is thermodynamically less severe.

These parametrically generated aerodynamic heating rate data were used for thermal analysis of the various candidate insulation systems. Radiation equilibrium temperatures for emissivity, $\varepsilon = 0.85$, are based on the following:

- Leading-edge stagnation heating rates peak at $M = 16.4$, altitude = 59.741 m (196,000 ft)
- Upper wing surface temperatures peak at $M = 6.4$, altitude = 26.36 m (86,500 ft)
- Lower wing surface heating rates and temperatures peak at $M = 7.9$, altitude = 35,537 m (116,000 ft)

Isotherms of the peak surface temperatures for upper and lower surfaces (excluding engine inlet interference effects) for the Star-Raker are shown in Fig. 23. Star-Raker lower-surface temperatures are from 222°C (400°F) to 333°C (600°F) lower than the orbiter due to lower reentry wing loading, 112 kg/m$^2$ versus 327 kg/m$^2$ (23 versus 67 lb/ft$^2$).

Typical variations of heat leak rate (W/m$^2$) into propellant filled tanks as a function of HRSI tile thickness for typical upper and lower wing tank locations are shown in Fig. 24. Total heat flux corresponds to a 1500-sec time period during ascent from $M = 0.8$ at 10,868 m (35,000-ft) altitude.

![Figure 23. Isotherms of Peak Surface Temperatures During Ascent](image)

![Figure 24. Heat Leak Into Propellant-Filled Wing Tanks](image)
Variation of bondline temperatures versus tile maximum temperature to thickness ratio is shown in Fig. 25. Curve A relates dry tank data for nominal upper surface positions. Curve B, shows LiH propellant-filled lower surface conditions, and Curve C shows the effects of cold ullage gas in contact with wing upper surface.

Typical thermal response as a function of launch trajectory exposure time of insulation and wing tank structure is summarized in Fig. 26. Part (a) of the figure shows zero heat flux at the HRSI surface and the re-radiated component; Part (b) shows tile temperatures from the ceramic surface to the bondline; Part (c) summarizes structure temperatures from the foam bondline to the pressure vessel wall; and Part (d) plots heat flux rate and total flux.

![Figure 25. Variation of Bondline Temperature With Tile Thickness](image)

![Figure 26. Insulation Thermal Response](image)

![Figure 27. HRSI Tile Thickness Contours for 1770°C Bondline Temperature](image)

On the titanium-aluminum systems, which show promise for high-temperature applications currently being developed by the Air Force. For temperatures higher than 871°C, it is anticipated that an alloy will be available from the dispersion-strengthened superalloys currently being developed for use in gas turbine engines. Flexible supports are designed to accommodate longitudinal thermal expansion while retaining sufficient stiffness to transmit surface pressure loads to the primary structure. Also prominent in metallic TPS designs are expansion joints which must absorb longitudinal thermal growth of the radiative surface, and simultaneously prevent the ingress of hot boundary layer gases to the panel interior. The insulation consists of flexible thermal blankets, often encapsulated in foil material to prevent moisture absorption. The insulation protects the primary load-carrying structure from the high external temperature.

During the past two years, Rockwell and Pratt and Whitney Aircraft have participated in an Air Force Materials Laboratory sponsored program, F31615-75-C-1167, directed toward the exploitation of Ti3Al base alloy systems. The titanium aluminate intermetallic compounds based on the compositions Ti3Al (α2) and TiAl (λ), which form the binary Ti-Al alloys, have been shown to have attractive elevated temperature strength and high modulus/density ratios (Fig. 28).

![Figure 28. Tensile Properties for Ti-13.5Al-21.5Cb](image)

The titanium hardware depicted in Fig. 29 is an example of complex configurations that have been developed by utilizing a process which combines superplastic forming and diffusion bonding (SP/DB). This Rockwell proprietary process has profound implications for titanium fabrication technology, per se. In addition, the unprecedented low-cost hardware it generates promises to revolutionize the design of airframe structure. The versatile nature of the process is apparent, showing complex deep-drawn structure and sandwich structure with various core
configurations. This manufacturing method and the design freedom it affords offer a solution to the high cost of aircraft structure. Manufacturing feasibility and the cost and weight savings potential of these processes have been established through both IR&D efforts at Rockwell and Air Force contracts. These structures may be used for engine cowling, landing gear doors, etc., in addition to providing major TPS components.

Unit masses of the Star-Raker TPS concept, state-of-the-art TPS hardware, and advanced thermal-structural designs are compared with the unit masses of the orbiter RSI in Fig. 30. The unit mass of the RSI includes the tiles, the strain isolator pad, and the bonding material. The shaded region shown for the RSI mass is indicative of insulation thickness variations necessary to maintain the moldline over the bottom surface of the orbiter. The RSI is required to prevent the primary structure temperature from exceeding 177°C. The unit masses of the metallic TPS are plotted at their corresponding maximum use temperatures. The advanced designs are seen to be competitive with the directly bonded RSI.

Structural Analysis

The multi-cell wing tanks provide a structure which is capable of sustaining pressure while, at the same time, reacting aerodynamic loads. The tanks are sized based on ullage pressures of 221 to 234 kPa, absolute (32 to 34 psia) (LH₂), and 138 to 152 kPa, absolute (20 to 22 psia) (LO₂). Maximum wing-bending occurs at about Mach 1.2. The LH₂ and LO₂ wing tanks are the major load path for reacting these loads. The wing also supports the airbreather engine system.

The primary wing attachment is to the cargo bay structure. The cargo bay aft section, in turn, is connected to the LH₂ tank. The LH₂ tank interconnects the cargo bay, aft portions of the wing, the vertical surface, and the rocket engine thrust structure.

An ultimate factor of safety of 1.50 was used in the analysis. The prime driver in the structural sizing of the multi-cell wing tanks is the bending moment resulting from air loads at Mach 1.2. The net bending moment on the wing is the difference between the lift moment and the relieving moment due to LO₂ remaining in the wing. Trades were performed to determine the structural wing weights required to sustain these bending moments plus internal pressure (Fig. 31). An intermediate location was chosen for LO₂ propellant where lift moment was approximately two times relieving moment. Chordwise location of the LO₂ tank is about the wing c.g. line from the wing root to the wing tip LH₂ ullage tanks. LH₂ tanks are located forward and aft of the LO₂ tanks. A fuel transfer system similar to the British/French Concorde maintains propellant c.g. location during launch.

The wing tank was designed to sustain the loads from both internal pressure and wing bending. AL 2219-T87 was chosen for the tank material on the basis of high strength at cryogenic temperatures, fracture toughness, and weldability. Loads resulting from wing bending moments are dominant in determining membrane thickness, which is based on a maximum tank ullage pressure of 234 kPa, absolute, and an ultimate factor of safety of 1.50. Fig. 32 shows material thickness versus wing station due to pressure and wing bending. The column showing bending-only relates to the wing-bending contribution, not an unpressurized wing design.

The fuselage LH₂ tank is the primary load path for reacting total vehicle mass inertias due to rocket thrust during the maximum acceleration condition (3.0 g). Approximately 27% of

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Figure 29. Typical Titanium Aluminide, Superplastic Formed, Diffusion-Bonded Structures

Figure 30. Unit Mass of TPS Designs

Figure 31. Tank-Relieving Moments Versus LH₂/LO₂ Location

Figure 32. Material Thickness Versus Wing Station
the propellant remains at that time. The tank shown in Fig. 33 has a twin-cone "Siamese" configuration which is required for it to fit in the fuselage at maximum propellant volume. The forward end of the tank is cylindrical, while the aft end is closed out with a double modified-ellipsoidal shell. The bulkheads react the internal pressures whereas the sidewall carries pressure and axial compression loads. The bulkheads are monocoque construction, while the sidewall is an integral skin-stringer with ring frame construction. Tank configuration and bulkhead membrane and sidewall "amended" thickness requirements to sustain the internal pressure and axial compression loads are reported in Reference 1. The structural design of all cryo tanks is based on cryogenic temperature material properties and allowances.

Figure 33. Aft Centerline LH2 Tank Material Thickness

Mass Properties

Star-Raker mass properties are dominated by the tridelta wing structure, the thermal protection system, and the airbreather and rocket propulsion system. Table 2 is the final reference vehicle weight summary for a due-east 28.5° launch from KSC.

<table>
<thead>
<tr>
<th>Item Description</th>
<th>Reference Vehicle (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Airframe, aerosurfaces, tanks, and TPS</td>
<td>157.829</td>
</tr>
<tr>
<td>Landing gear</td>
<td>12.565</td>
</tr>
<tr>
<td>Airbreather propulsion</td>
<td>35.523</td>
</tr>
<tr>
<td>RCS propulsion</td>
<td>4.536</td>
</tr>
<tr>
<td>OMS Propulsion</td>
<td>2.268</td>
</tr>
<tr>
<td>Other systems</td>
<td>17.146</td>
</tr>
<tr>
<td>Subtotal</td>
<td>300.370</td>
</tr>
<tr>
<td>10% growth</td>
<td>30.037</td>
</tr>
<tr>
<td>Total inert weight (dry weight)</td>
<td>330.407</td>
</tr>
<tr>
<td>Useful load (fluids, reserves, etc.)</td>
<td>21.500</td>
</tr>
<tr>
<td>Inert weight and useful load</td>
<td>351.907</td>
</tr>
<tr>
<td>Payload weight</td>
<td>89.167</td>
</tr>
<tr>
<td>Orbital insertion weight</td>
<td>441.073</td>
</tr>
<tr>
<td>Propellant ascent</td>
<td>1.826,890</td>
</tr>
<tr>
<td>GLOW (post jettison launch gear)</td>
<td>2,267.960</td>
</tr>
</tbody>
</table>

Table 2. Weight Summary, KSC Launch

Summary and Conclusions

The combined systems design/performace and technology development studies in FY 1978 produced a number of significant results.

1. Demonstrated, with end-to-end simulation, the ability of the vehicle to take off from KSC, cruise to the equatorial plane, insert into a 557-km (340-nm) equatorial orbit with a 57,200-kg (126,200-lb) payload and then to reenter and return to the launch site; also demonstrated the ability to deliver a 89,167-kg (196,580-lb) payload with a due-east launch.

2. Devised a multi-cycle airbreathing engine system concept for operation in turbfan, air-turbo-exchanger, and ramjet modes to provide an effective propulsion match with takeoff, cruise, and acceleration requirements.

3. Showed that Star-Raker lower surface temperatures during reentry are several hundred degrees lower than the lower surface temperatures of the orbiter because of a lower wing loading. As a result, an advanced titanium aluminum system shows promise of being lighter than the RSI tile system for Star-Raker application.

4. Performance was found to scale almost linearly between 4.53,590 kg (1,000,000 lb) and 2,267,960 kg (5,000,000 lb) GLOW vehicle systems, provided that consistent wing loading and thrust loading are used.

5. Nose loader vehicles can utilize existing cargo-handling techniques used for commercial 747 or military C-5A aircraft systems modified for space at significant weight and cost savings over top-loading vehicles.

6. Aircraft-type operations are significantly more effective and less costly than vertical launch single- or two-stage systems. They have a potential of $22 to $33 per kilogram ($10 to $15 per pound) of payload in orbit.

7. Powered landing capability at any commercial or military airport currently handling 747 or C-5A aircraft.

8. The principal results of the study, which had been performed to an accelerated schedule, were briefed to NASA/MSFC in January 1978; and technical issues resulting from NASA evaluation of the data were essentially resolved by mid-year. Results of the Star-Raker study were utilized in the SPS contract (NASA-32475). Adjustments were included in the SPS final report. Adjusted results were subsequently utilized in the SPS contract extension. As an outgrowth of the Rockwell and NASA/MSFC study and evaluation activities, NASA/MSFC was later to conduct a contract (NFT-01-602-099), "Airbreathing Engine/Rocket Trajectory Optimization," to the University of Alabama.

9. Suborbital transportation of payloads between point-to-point earth surface launch/landing sites is available commercially as a fallout from space-oriented missions.

10. Key technical areas for further analysis and research are: advanced airbreathing propulsion system; metallic TPS systems; wet-wing tankage and structures analyses; and propellant utilization and c.g. control system analyses.

Appendix

Star-Raker/RASV Comparison (Mini Star-Raker)

Performance trades were developed and comparisons made for (1) a 544,310-kg (1,200,000-lb) version of the Star-Raker with its airbreather/rocket (AB/R) propulsion and (2) the Reusable Aero
dynamic Space Vehicle (RASV) with its all-rocket propulsion, as
defined by Boeing. The study was in response to an expression of interest by government agencies who had sponsored the RASV study. The objectives were to determine whether the AB/R configuration could achieve specific performance goals, to identify key operational and performance differences between the AB/R and all-rocket vehicles, and to determine conditions (crossover) at which performance gains are realized.

Vehicle configurations, major design characteristics, and performance ground rules to permit comparisons of useful payloads are summarized in Fig. 34. The scaled-down AB/R Star-Raker has a tridelta wing planform and a Whitcomb airfoil section similar to the Star-Raker designed for the SPS mission. Holding the same gross lift-off weight (GLOW) for the comparison was a study ground rule. The higher empty weight for the scaled down mini Star-Raker represents the addition of airbreathing engine system and net propellant tank weight increments to the basic RASV weight. As noted, the RASV has a 4536-kg (10,000-lb) payload capability to a 185-km (100-nmi) polar orbit and a 11,340-kg (25,000-lb) capability to a 185-km (100-nmi) orbit at 280°. The payload capability of the mini Star-Raker was to be defined parametrically.

Optimal trajectories for HTO/SSTO concepts, using combined airbreathing rocket (AB/R) and all-rocket propulsion systems were determined as reported in Reference 4. Approximately 1500 seconds of AB operation are required for the AB/R system to reach a transitional velocity (VAB/R) of 1830 m/s (6000 ft/s), while 300 seconds are required by the rocket system to reach the same velocity. After VAB/R, the time increment required to reach orbital velocity is the same for both vehicles. The corresponding aerodynamic heating parameters, which are proportional to total enthalpy flux histories, were also determined as reported in Reference 4. The data show that the AB/R system will be exposed to a higher heating environment during launch than the all-rocket system because the AB/R system must fly at lower altitudes where the AB engine operates most efficiently.

Total throw weight as a function of launch wing loading and AB/R transitional velocity (VAB/R) are also presented in Reference 4. For a given transitional velocity, the throw weight increases rapidly between 4788 and 7182 N/m² (100 and 150 lb/ft²) wing loading, but flattens out between 7182 and 9576 N/m² (150 and 200 lb/ft²). Therefore, considering the substantial increase in the aerothermal parameter between 7182 and 9576 N/m² (150 and 200 lb/ft²), a wing loading of 7182 N/m² (150 lb/ft²) was selected for the rest of the trade studies.

The useful payload capability for the AB/R system, which considers the added weight penalty resulting from AB engines and increased tank weights, is shown parametrically in Fig. 35. For the AB/R configuration, the minimum payload requirement of 4536 kg (10,000 lb) to a polar orbit can be met with a VAB/R/(T/Wg)AB combination of 1220/10, 1372/8, or 1830/6. The payload requirement for 13,608 kg (30,000 lb) into the 280° inclined orbit is also exceeded by these engine performance/weight combinations. Fig. 35 also shows that the payload capability of advanced technology AB engines of Mach 6 and (T/Wg)AB = 8 are almost double those of a vehicle with state-of-the-art 177,930-N (40,000-lb) thrust Mach 4 engines (8 ≤ (T/Wg)AB ≤ 10). However, (T/Wg)AB will tend to decrease as VAB/R increases due to the added weight incurred by engine and inlet design complexities resulting from aerothermodynamic effects.

In summary, the performance trade studies for the mini Star-Raker show the following:

1. The mini Star-Raker is capable of placing the desired 4536 and 13,608 kg (10,000 and 30,000 lb) of useful payload into 185.2-km (100-nmi) polar and 280° orbits, respectively.
2. These payload capabilities can be achieved with various combinations of VAB/R ranging from 1219 to 1829 m/s (4000 to 6000 ft/s) and (T/Wg)AB ranging from 6 to 10.
3. The optimal wing loading is approximately 7182 N/m² (150 lb/ft²) when performance and aerothermodynamic effects are taken into account.
4. The mini Star-Raker is competitive from a performance standpoint with the all-rocket RASV system with moderate advancements in AB propulsion system technology (T/Wg)AB = 8-10, and VAB/R = 1219 m/s.
5. The mini Star-Raker offers the potential of major growth in payload capability, e.g., ~11,340 kg (25,000 lb) into polar orbit and ~22,680 kg (50,000 lb) into 280° orbit, with significant AB propulsion system technology advancement to VAB/R = 1829 m/s (6000 ft/s) and (T/Wg)AB > 8.
6. A major advantage of the AB/R system is operational flexibility. Particularly, the atmospheric cruise capability with AB engines makes this vehicle an excellent air transportation system for ferrying and payload pick-up missions. This capability results in reduced ground and auxiliary air transportation system requirements for logistics support. The mini Star-Raker without LO2, which is required for space operation only, weighs approximately 136,080 kg (300,000 lb) [W/S ≤ 1915 N/m² (40 lb/ft²)] including the payload and cruise fuel for 6440 km (4000 statute miles). Therefore, the mini Star-Raker can operate in this mode from any airfield which can handle the C-5A or the Boeing 747 and which adds LH2 facilities.

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References


