Table 1 Flowfield properties

Region	Turning angle, deg, $\delta_{(n-1)-n}$	Shock angle, deg, $\theta_{(n-1)-n}$	Flow direction, deg	Mach Number	Static pressure ratio, p/p_0	Static temp. ratio T/T_0	Total temp. ratio, T_t/T_{t0}
0	0	0	0	5	1.0	1.0	1.0
1	10	19.376	+10.000	3.999	4.392	1.429	1.0
2	18	30.239	−8.000	2.172	13.896	2.428	1.0
3	26.644	5.683	+18.644	1.870	6.375	7.082	2.006
4	18.644	28.266	+18.644	3.151	6.375	2.010	1.0
5	18.644	14.267	0	1.840	3.466	9.744	2.724
6	1.184	19.312	+17.460	3.086	7.011	2.065	1.0
7	17.460	6.662	0	2.447	3.446	3.891	1.425

The mass flow ratio (i.e., the split) for the streams is obtained from the geometry:

$$\frac{Y_0'}{Y_0} = \left[\frac{\sin(\theta_{0-1} - \delta_{0-1})\sin(\theta_{1-2} - \delta_{1-2})}{\sin\theta_{0-1}\sin\theta_{1-2}\sin\theta_{2-3}}\right] \left[\cos(\theta_{2-3} + \delta_{1-2} - \delta_{0-1}) + \frac{\cos(\theta_{3-5} + \delta_{3-5})\sin(\theta_{2-3} + \delta_{2-3})}{\sin\theta_{3-5}\sin\theta_{2-3}}\right] \Bigg/ \cot\theta_{0'-4} + \frac{\cos(\theta_{4-6} - \delta_{0'-4})\sin(\theta_{0'-4} - \delta_{0'-4})}{\sin\theta_{4-6}\sin\theta_{0'-4}}$$

where δ is the turning angle and θ is the shock (or OPH) angle measured from the upstream velocity vector. The double subscripts refer to the regions upstream and downstream of the shock or OPH corresponding to Fig. 2.

To enable the reader to check the example case, the flow properties in each of the regions are listed in Table 1. Note that there is considerably more heat addition in the primary streamtube, as indicated by the value of 2.724 for T_t/T_{t0} in region 5 vs 1.425 in region 7. In this example, the maximum internal contraction ratio, $A_0/A_d = 1.204$ (ratio of captured streamtube height to minimum internal duct height), occurs in the plane of the cowl lip. On the other hand, if the dividing streamline were replaced by a solid surface which would be required to obtain the same performance in the absence of the beneficial effect of thermal compression, the respective A_0/A_d values would be much greater, i.e., $A_0/A_d = 2.036$ in the primary, and $A_0/A_d = 3.012$ in the secondary.

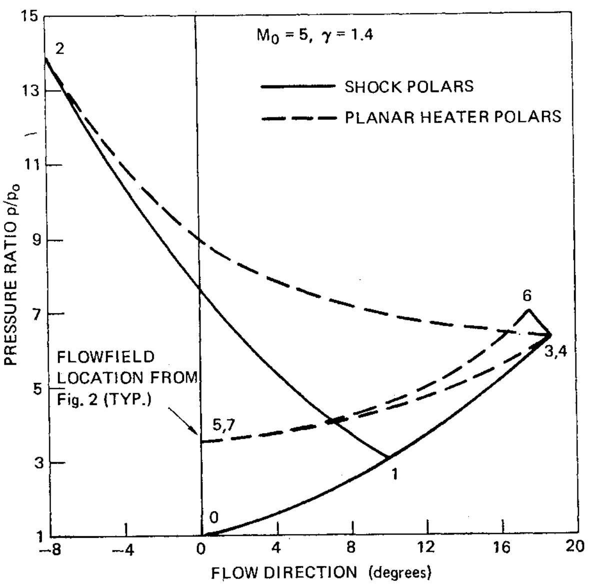

Fig. 3 Oblique planar heater and shock polars.

Consistent with the intent in the original paper,[1] the relatively simple gasdynamic models for thermal compression are used here to facilitate the basic understanding of the concept. The very formidable problem of developing a fuel distribution system for a realistic engine design that would place planar flame fronts in the desired locations is beyond the scope of this discussion.

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Aerospace Application of Atmospheric Rendezvous

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Introduction

THERE are a number of situations in which an atmospheric form of rendezvous has been used in aeronautics (see Fig. 1). Space rendezvous has also been used to increase the efficiency and flexibility of operations in the Apollo manned lunar mission. An examination of aerospace missions of present interest indicates that orbital logistics and hypersonic flight vehicles may benefit in efficiency and flexibility of operation by the use of an atmospheric rendezvous. The

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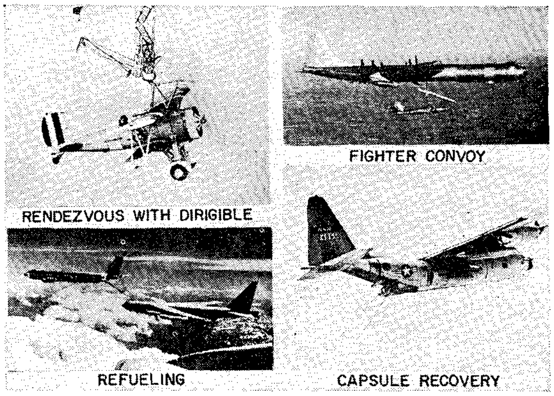

Fig. 1 Past experience with atmospheric rendezvous.

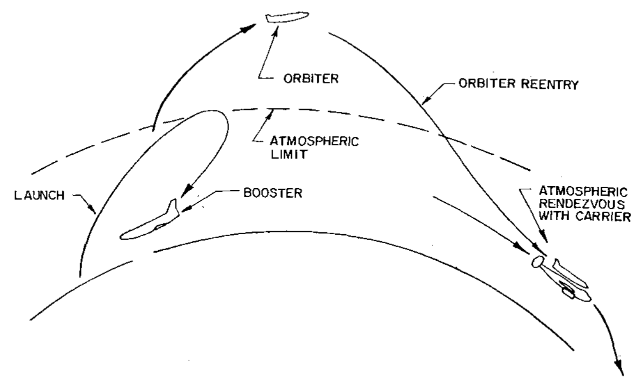

Fig. 2 Sequence of operations in atmospheric rendezvous concept of orbital logistics.

purpose of this paper is to summarize the results of a study[1] of the use of atmospheric rendezvous in aerospace missions and to show, generally, the feasibility and benefits to be derived from a rendezvous mode.

In the case of these transport missions, large changes in velocity, altitude, and atmospheric density are involved, and it is inefficient to carry elements to high speeds when lower speeds are sufficient. Application of the atmospheric rendezvous concept to two-stage recoverable launch vehicles would permit the in-flight or second stage, and in some cases—possibly the first stage, to be constructed without certain low-speed elements such as wings, landing gear, or subsonic ferry flight propulsion. These vehicles would be received in an atmospheric rendezvous by a carrier vehicle in the terminal portion of their flight and ferried to a suitable landing site. The reduction in weight of the booster or of the second stage of these vehicles would decrease the size of the launch vehicles for a given payload by a significant amount and give some greater flexibility in choice of landing sites.

Results and Discussion

The general mode of operation for orbital missions with the atmospheric rendezvous concept is shown in Fig. 2. Recovery of the orbiter can involve either docking of the orbiter to the carrier vehicle or acquisition of the orbiter in a towing mode. The rendezvous may be made at subsonic speeds by a large subsonic carrier aircraft, or at supersonic or hypersonic speeds by the use of the booster stage of the logistics vehicle.

An investigation of the feasibility of the atmospheric rendezvous operation requires defining reasonable configurations for the vehicles involved. Weight data were based on various space-shuttle studies.

In order to obtain good subsonic glide characteristics, the body was shaped to obtain a lift-to-drag ratio of about 4. The general arrangement of the orbiter vehicle design is shown in Fig. 3.

Estimated rendezvous weights of the orbiter from 175,000 to 210,000 pounds were obtained. A value of 200,000 lb, having approximately a 66,000 lb or 25% weight reduction compared to the landed weight of a conventional space-shuttle configuration, was selected for feasibility studies.

Mission weight data for the analyses made during this study are shown in Table 1. Results are given for the conventional landing mode and for the atmospheric rendezvous

Table 1 Mission Weight Data

Concept→	Conventional	Atmospheric Rendezvous		
Booster Recovery→	Landing	Landing	Landing	Airplane Tow
Orbiter Recovery→	Landing	Airplane	Booster	Airplane
Weights, pounds				
Landed				
Orbiter	265,000	200,000	200,000	200,000
Booster	540,000	418,000	528,000	341,000
Lift-off				
Orbiter	860,000	665,000	665,000	665,000
Booster	3,780,000	2,974,000	3,424,000	2,471,000
Total	4,640,000	3,639,000	4,089,000	3,136,000

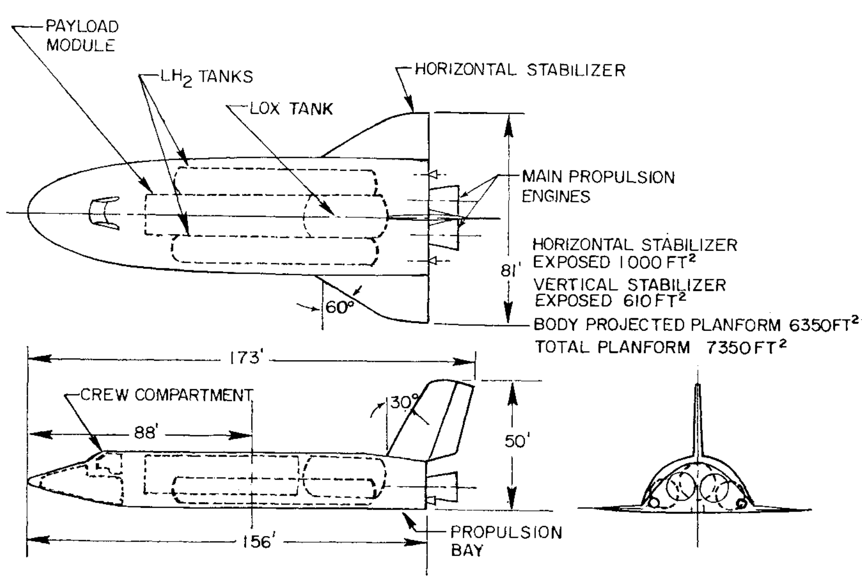

Fig. 3 An atmospheric rendezvous orbiter design.

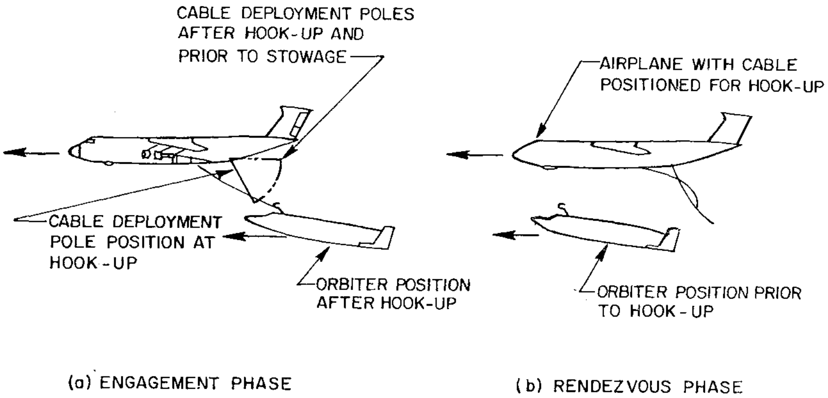

Fig. 4 Recovery of orbiter by towing mode.

mode with several options. It is evident that significant reductions in gross lift-off weight are possible for concepts that include orbiter-airplane or booster-airplane rendezvous. The decrease in gross lift-off weight for orbiter-booster rendezvous is relatively small because of the increased booster-wing area and cruise-propulsion capability required.

Characteristics of both orbiter-airplane and orbiter-booster rendezvous have been studied. Although the orbiter-booster case appears to be feasible, several disadvantages have been identified. The following paragraphs emphasize a subsonic orbiter-airplane rendezvous because it appears to be the more promising approach. A modified Lockheed C-5A was selected as the recovery airplane for the study, primarily because of its size and pay load capacity of 265,000 lb.

A substantial capability for the orbiter and airplane to compensate for errors exists. The orbiter can maneuver laterally up to 2000 miles during re-entry, and longitudinally up to 360 naut miles from a speed of 10,000 fps. The C-5 carrier can travel up to 1400 naut miles to a rendezvous, and compensate in range as much as 3 naut miles in the last 2 min before rendezvous.

A study was made to determine rendezvous guidance characteristics and requirements for the case of orbiter-airplane rendezvous at subsonic speed. There are large variations of flight conditions and guidance activities between orbiter re-entry and completion of rendezvous. During the final 3 min all rendezvous maneuvers are to be performed by the airplane. Conditions at 2 minutes to-go in terms of relative altitude and altitude rate are very similar to those for approach to a lunar landing. The Apollo lunar module uses a landing radar to measure altitude and altitude rate, and controls attitude and thrust as functions of these inputs. The rendezvous phase could be completed at an altitude of 32,000 ft, with the orbiter above and within 50 ft of the airplane, and with relative velocity less than 5 fps. Rendezvous guidance requirements can be satisfied with existing inertial navigation, communication, and radar equipment.

Recovery of the orbiter by the towing mode is illustrated in Fig. 4. The airplane approaches the orbiter at a small relative velocity from the rear and above. The tow cable, supported by controllable deployment poles, engages a hook on the upper surface of the orbiter. The operation is similar to the air snatch of a small re-entry pay load during parachute descent, except that the relative velocity is much smaller, and the recovered weight is much larger. As previously discussed, further reduction in booster size can be realized if the booster is recovered by an airplane in the tow mode. This operation permits removal of booster cruise engines and cruise fuel, and a subsequent reduction in fuel required for launch (see also Ref. 2). The airplane can tow the booster if two additional engines are added.

A conceptual design of a system to dock the orbiter to the airplane is shown in Fig. 5. The main component of the system would be a telescoping boom extending through the upper portion of the airplane fuselage. It supports a latching mechanism that engages a docking cone at the lower surface of the orbiter. Initial acquisition would be achieved with significant separation between vehicles to minimize aerodynamic interference. This initial operation is similar to that for air-to-air refueling, as shown in Fig. 1. The docking concept appears to provide more favorable relative vehicle positions than air-to-air refueling from the standpoint of wake interference. Required airplane modifications (in addition to the docking system) include a high-drag device to permit matching the orbiter's glide characteristics, and probably a new tail configuration for satisfactory effectiveness with the orbiter docked.

The time available to perform a docking is shown in Fig. 6. Relative motion data and rendezvous guidance studies indicate that the airplane and orbiter vehicles can be positioned to begin docking at an altitude of 32,000 ft. Thirty seconds are allowed for engaging the latching mechanism on the main telescoping boom. This period is reasonable, based on

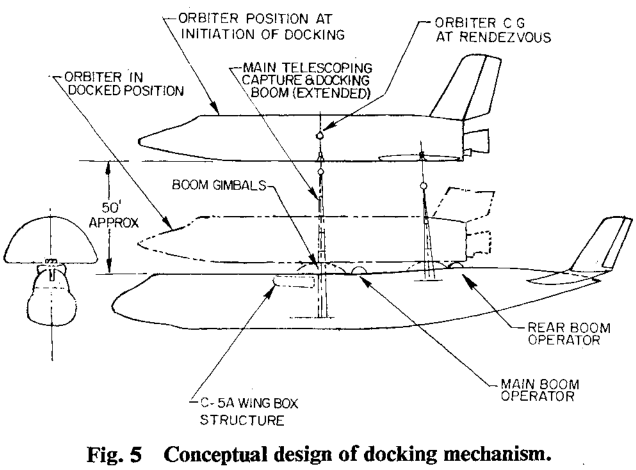

Fig. 5 Conceptual design of docking mechanism.

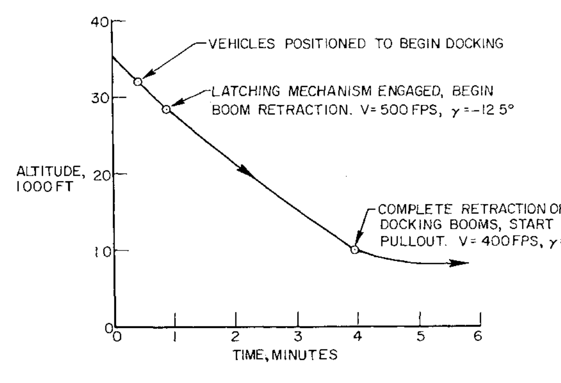

Fig. 6 Docking sequence.

modern air-to-air refueling operations. An additional 3 min are available before the vehicles descent to an altitude of 10,000 ft, which is considered a safe altitude for completion of docking and pulling out of the glide.

Concluding Remarks

An investigation has been made of the general operational and design considerations for an atmospheric rendezvous mode for orbital logistics vehicles that uses fully recoverable vehicle elements. It is concluded that this mode of operation is feasible, and that it would result in reductions in the size of the orbiter and booster stages of the launch vehicle, provide a powered landing with go-around capability for every mission, and achieve additional lateral range performance. A subsonic rendezvous in either a towing or docking mode seems more attractive than the supersonic case. Recovery of the booster stage in a subsonic towing mode seems to offer some advantage in weight reduction and flexibility of operation.

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Current Drainage to a High Voltage Probe in a Dilute Plasma

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Introduction

SOLAR cell arrays in the kv range are being proposed for power generation on satellites. If the solar cell cover glass or the dielectric material insulating the connecting tabs between solar cells develops holes or cracks, a large drainage current from the ambient plasma may ensue. Cole et al.[1] measured drainage current on the order of milliamps through 0.0254 cm diam. holes in Kapton H polymide films biased up to 3000 v in a 8 km/sec plasma. This paper reports results for electron drainage current through known size holes in three dielectric materials immersed in a 4 km/sec plasma stream. The dielectric materials were a quartz solar cell cover glass, Kapton H polyimide film, and fluorinated ethylene propelene (FEP) type C. The experimental results are compared with results calculated using a computer program developed by Parker.[2–5]

Facility

The dielectrics were placed in a pyrex bell jar (45.7 cm in diam, 76.2 cm long) attached to the side of a large vacuum tank which houses a Kaufman ion thruster generating an argon plasma (Fig. 1). The vacuum tank pressure was approximately 2×10^{-5} torr. The test specimens were mounted on one end of a 3.2 cm-diam. cylindrical pyrex tube. The ammeter used to measure the drainage current was located between the high voltage power supply and the test specimen. The electrical connecting lines and the ammeter were shielded, with the shield at the potential of the test specimen. In all cases the high voltage probe was biased positively with respect to ground.

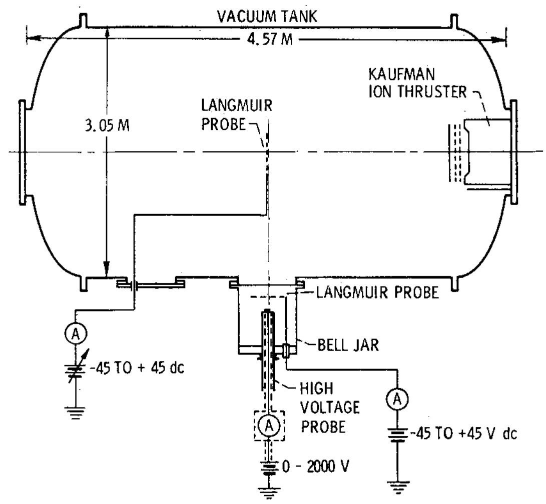

Fig. 1 Sketch of experimental facility (not to scale).

Plasma Diagnostic

Two cylindrical tungsten Langmuir probes and a Faraday cup were used to diagnose the plasma. One of the Langmuir probes (25.4 cm long by 0.0254 cm in diameter) was located on the centerline of the tank at the axial station of the port to the bell jar. The other Langmuir probe (12.4 cm long by 0.0127 cm in diameter) was located in the bell jar approximately 20.32 cm in front of the testing location.

Velocity

On a plane perpendicular to the axis of the tank at the axial position of the bell jar, the ion number density in the ion engine exhaust beam is larger than that in the bell jar. The largest ion density is on the centerline of the beam. A potential difference exists between the engine exhaust beam and the inside of the bell jar. This potential difference accelerates the charge exchange ions in the beam radially. Some of these enter the bell jar along with other scattered ions. The velocity of the ions was calculated from

$$V = (2e\Delta v/M)^{1/2} \tag{1}$$

where Δv is the potential difference between the centerline of the ion engine plasma beam and the plasma potential (assumed constant) in the bell jar determined from the Langmuir probe measurements, and M the mass of the ions.

Number density

The plasma number density in the bell jar was found from $I - v$ characteristic of the Langmuir probe located in the bell jar. For the electron densities expected in the bell jar and the beam, the Debye length is greater than the probe diameter, therefore, a thick sheath is expected about the Langmuir probe. According to electric probe theory,[6] when there is a thick sheath surrounding cylindrical electric probes operating in the electron current saturated region and $ev \gg kT$, the square of the current should vary linearly with the applied

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